#15

5/19/03

Received May 19 2003 TC 3700

Attorney Docket No. 3600.100 Cont.

IN THE UNITED STATES PATENT AND TRADEMARK OFFICE

In re Reissue Application of	)	Examiner: C. Verdier
DAVID A. SPEAR ET AL.	)	Group Art Unit: 3745
Appln. No.: 09/874,931	)	
Filed: June 5, 2001	)	
For: SWEPT TURBOMACHINERY BLADE	)	Application to reissue U.S. Patent 5,642,985

Commissioner for Patents
P.O. Box 1450
Alexandria, VA 22313-1450

DECLARATION OF FRANS A.E. BREUGELMANS

Sir:

I, Frans A.E. Breugelmans, do hereby declare as follows:

1. In 1962 I received from Leuven University in Belgium the degree of Master of Science in Mechanical Engineering (also known as a Civil Engineer degree). In 1963 I completed post-graduate studies at the Training Center for Experimental Aerodynamics, which in 1964 was renamed the Von Karman Institute ("VKI"). VKI is a non-profit educational and scientific organization with departments in aeronautics and aerospace, environmental and applied fluid mechanics, and turbomachinery and propulsion. See http://www.vki.ac.be. From 1963 to 1976, I lectured and performed research and development in the VKI turbomachinery and propulsion department. In 1976 and 1977 I was a visiting associate professor in the Mechanical Engineering Department of Iowa State University, Ames, Iowa, lecturing on the physics and aerodynamics of the flow in gas turbine engine compressors and fans. In 1977 I returned to VKI as a full professor in the turbomachinery and propulsion department. From 1977 to 1997 I was

the head of that department, and from 1991 to 2000 I was the assistant director of VKI. I retired from VKI in 2000, and I am currently an honorary professor in the VKI turbomachinery and propulsion department. During my professional career, up to and including the present, I have also been involved with numerous educational and professional organizations other than VKI, as a result of my recognized expertise in gas turbine engine compressors and fans. My *Curriculum Vitae*, attached hereto as Exhibit 1, lists the professional positions and memberships I hold or have held.

2. At VKI, I have designed numerous gas turbine engine components, including compressors and fans, for research and development projects sponsored by private industry and governmental organizations. These include the design, construction, and testing of supersonic compressors for the United States Air Force, developing blade sections for transonic and supersonic fans, and compressor stall studies, to name just a few. A more complete list is included in my *Curriculum Vitae*.

3. I am named as an author on numerous publications and project reports in the field of gas turbine engines, most of which publications and reports relate to the physics and aerodynamics of the flow in axial-flow compressors and fans of compressors. A list of my publications is included in my *Curriculum Vitae* (Exhibit 1). I am named as an inventor on U.S. Patent Appln. Pub. No. 2002/0066267 A1.

4. As a result of my education, training, and experience, I have expertise in the analysis of the flow through axial-flow turbomachinery, particularly compressors and fans, for gas turbine engines, and in the physics and aerodynamics of the interaction of gas turbine engine working medium gases with the blades and ducts of axial-flow fans for gas turbine engines.

5. The subject matter discussed in this declaration involves complex, supersonic fluid flow through the fan of a gas turbine turbofan engine. Knowledge of some basic principles of jet

propulsion and fluid mechanics is required to understand the topics discussed herein. To that end, attached Exhibit 2 explains some of the fundamentals of those subjects.

6. In connection with this declaration I have studied U.S. Patent No. 5,642,985 to David A. Spear et al. ("the UTC '985 Patent") (attached as Exhibit 3), reissue of which I am informed is sought by the above-identified present application ("the UTC Reissue Application"). On information and belief, claims 18, 19, 22, and 23 set forth elsewhere herein are to be presented in the UTC Reissue Application. I have also studied the patent and literature references listed on the Information Disclosure Citation forms attached hereto as Exhibit 4.

7. It is my understanding that the UTC Reissue Application was filed to initiate an interference in the U.S. Patent and Trademark Office with U.S. Patent No. 6,071,077 to Paul A. Rowlands, which names as its assignee Rolls-Royce PLC ("the Rolls '077 Patent") (attached as Exhibit 5). It is also my understanding that the UTC Reissue Application has the same disclosure as the UTC '985 Patent. I have studied the Rolls '077 Patent in connection with this declaration.

The Common Invention Of The UTC '985 Patent And The Rolls '077 Patent

8. The claims in the present application and their relation to the claims in the Rolls '077 Patent are discussed in detail below. However, I believe the discussion in that regard will be better understood by first considering in general the subject matter of the UTC '985 Patent, which is the precursor to the present application, and the Rolls '077 Patent. (Technical terms and concepts in the following discussion are explained more fully in Exhibit 2.)

9. The Rolls '077 Patent and the UTC '985 Patent both relate to a fan stage of a ducted fan gas turbine engine, and more particularly to an advantageous blade configuration for such a fan. The fan shown in Figures 1 and 2 of the UTC '985 Patent has multiple blades disposed in a duct. UTC '985 Patent, col. 2, lines 42-44 and 56-58. The blades are swept to minimize the

adverse effects of supersonic flow of the working medium over the blades. UTC '985 Patent, col. 1, lines 4-7. The Rolls '077 Patent also relates to a ducted fan gas turbine engine having swept fan blades. Rolls '077 Patent, col. 1, lines 7-10.

10. The fans in both patents rotate at speeds that produce supersonic flow over the fan blades at some point along the blade span. UTC '985 Patent, col. 3, lines 6-10; Rolls '077 Patent, col. 1, lines 31-33. This supersonic flow causes the formation of shock waves and reduces the efficiency of the fan. UTC '985 Patent, col. 3, lines 10-14; Rolls '077 Patent, col. 1, lines 38-45. It was well known prior to 1995 that sweeping the leading edge of a supersonic fan blade will mitigate these losses. UTC '985 Patent, col. 1, lines 27-31; Rolls '077 Patent, col. 3, lines 11-20 and 41-45. (The terms "sweep" and "sweep angle" have accepted definitions, used in both the UTC '985 Patent, col. 3, lines 20-29, and the Rolls '077 Patent, col. 2, lines 5-10).

11. To that end, each of the UTC and Rolls fan blades has a leading edge with forward sweep in an inner region followed by a rearward swept intermediate region, as seen in Figure 2 of the UTC '985 Patent and Figure 5a of the Rolls '077 Patent. As noted in paragraph 10, the rearward sweep in the intermediate region is important because it mitigates shock losses and thus increases the fan's efficiency, by reducing the Mach number of the supersonic airflow in a direction perpendicular to the blades' leading edges. UTC '985 Patent, col. 3, lines 14-18; Rolls '077 Patent, col. 3, lines 14-17. Forward sweep in the inner region of a blade had been used previously to make a blade with rearward sweep further out along the blade (that is, in the intermediate region) practicable from the standpoint of the mechanical stresses on the blade. Exhibit 2, paras. 23-26; see also U.S. Patent 4,012,172 to Schwaar et al. ("Schwaar") (attached as Exhibit 6) and U.S. Patent 4,726,737 to Weingold et al. ("Weingold") (attached as Exhibit 7). The Rolls '077 Patent at column 3, lines 30-33, alludes to the mechanical reasons for using forward sweep in the inner region.

12. Accordingly, the forward-rearward swept blade geometry described thus far (a leading edge swept forward in an inner region and swept rearward outwardly therefrom) was known before the filing dates of the applications that became the UTC '985 and Rolls '077 Patents. In addition, the prior art recognized the difficulties in using a blade with a rearward swept leading edge, and proposed numerous approaches for overcoming those difficulties and maximizing the advantages of operating gas turbine engine fan blades in a supersonic flow regime. Those working in the art were aware that supersonic fan blades should have performed better with a rearward swept region radially outward of a forward swept inner region, but no such blade ever gained wide acceptance. Schwaar, Weingold, and U.S. Patent 3,989,406 to Bliss ("Bliss") (attached as Exhibit 8) proposed improving the aerodynamic performance of such a blade by providing a high degree of rearward sweep to render subsonic the flow perpendicular to the leading edge, thereby eliminating shocks altogether. Those solutions were impracticable. Exhibit 2, paras. 23-26.

13. A principal problem with prior forward-rearward swept blades is that a shock can form in front (that is, upstream) of the blade in an outer tip region near the duct wall. UTC '985 Patent, Figs. 2 and 3 (endwall shock 64, see also col. 3, lines 30-40); Rolls '077 Patent, Fig. 3a (showing the shock wave 36 in front of the blade in the tip region). The UTC '985 Patent and the Rolls '077 Patent provide slightly different depictions of the shock system created by such a blade, but both point out that it can include a shock upstream of the blade in the tip region. Such a shock results in a decrease in operational efficiency, and can cause heavy vibration and fan stalling and surge, which requires corrective action to overcome in cases where the stalled fan fails to provide sufficient airflow for engine operation. UTC '985 Patent, col. 3, lines 55-57; Rolls '077 Patent, col. 3, lines 41-45, col. 4, lines 11-17.

14. The same feature of the blade in the UTC '985 Patent and the Rolls '077 Patent solves that problem: a blade leading edge outer region is translated forward relative to the more highly swept leading edge at the end of the intermediate region, thereby preventing the shock from forming ahead of the leading edge. UTC '985 Patent, col. 4, line 63, to col. 5, line 4; Rolls '077 Patent, col. 4, lines 18-21, and col. 7, lines 28-30.

15. The UTC '985 Patent explains that the shock system includes an endwall shock portion adjacent the duct wall and a passage shock portion across the passages between the fan blades. See col. 3, lines 30-57, and col. 4, lines 37-44. The passage shock portion extends across each interblade flow passage, col. 3, lines 50-52, col. 4, lines 42-44, and Fig. 3, because the blade leading edge's intermediate region travels faster than the speed of sound (M > 1) relative to the incoming flow and creates a shock wave, as would be expected.

16. The UTC '985 Patent says that the shape of the blade influences the shock in the endwall region, adjacent the duct wall, because pressure waves generated by the blades are reflected by the wall. Under certain conditions, a shock wave forms ahead of each blade's leading edge. UTC '985 Patent, col. 3, lines 36-40, and Fig. 2 (endwall shock 64).

17. The Rolls '077 Patent likewise points to a shock 36 with a portion in each interblade passage, and also shows that the rearward swept leading edge at the blade outer region can cause another shock portion to form ahead of each blade's leading edge. The Rolls '077 Patent explains the origins of the shock slightly differently, col. 3, line 60, to col. 4, line 6 (which I believe was intended to refer to the depiction in Figs. 3a and 3b, rather than Figs. 4a and 4b), but both patents nonetheless describe virtually identical shock systems.

18. From the teachings in the Rolls '077 Patent and the UTC '985 Patent, one skilled in this art would have understood at the time the UTC and Rolls patent applications were filed, and would understand now, that both patents teach that the blade leading edge in the tip region must

be translated forward a sufficient distance to cause the blade to intercept the shock near the duct wall. UTC '985 Patent, col. 4, line 33, to col. 5, line 4; Rolls '077 Patent, col. 4, lines 18-21, and col. 7, lines 28-30. He or she would also have appreciated that those patents correctly teach that translating the tip region forward to reduce rearward sweep solved the problem of providing a practicable swept fan blade. See Exhibit 2, paras. 25-26.

19. That the Rolls '077 Patent explicitly calls for a forward swept leading edge in the blade outer region adds nothing to the teaching of the UTC '985 Patent. A fan engineer working at the time of the UTC '985 Patent's filing date obviously would have read it to teach the necessity of using a fan blade configuration that achieves endwall shock interception at all engine operating conditions. In my opinion the same fan engineer would have found that the prescription of the UTC '985 Patent, namely translating forward the leading edge of a forward-rearward swept blade in the outer region, inevitably leads to the use of the forward sweep called for by the Rolls '077 Patent. I believe that to be the case even under normal engine operating conditions. But it is especially true considering that at the time (as well as now) standard practice required fan operability not only under normal conditions, but also at the most severe operating conditions the engine would encounter. <u>See</u> Rolls '077 Patent, col. 1, lines 17-23. My expert opinion is that a fan engineer following the UTC '985 Patent's command to intercept the endwall shock under all engine operating conditions would have had to translate the leading edge of the blade's outer region sufficiently far forward to impart forward sweep as in the Rolls '077 Patent.

<u>UTC Claims 18 And 23 And Rolls Claims 1 And 8 Define The Same Invention</u>

20. UTC independent claim 18 and Rolls independent claim 8 are essentially broader in all respects relevant to this discussion than any other claim in the Rolls '077 Patent or the present

UTC Reissue Application. That is, all of the other claims in the Rolls '077 Patent and the present application contain additional features not found in these claims.

21. The following chart highlights the precise differences between UTC claim 18 and Rolls claim 8:

Present Application Claim 18	Rolls '077 Patent Claim 8
A fan stage of a ducted fan gas turbine engine that is rotatable about an axis of rotation and defines a downstream direction along the axis of rotation, comprising:	A fan stage of a ducted fan gas turbine engine that is **at least in part** rotatable about an axis of rotation and defines a downstream direction along the axis of rotation, comprising:
a fan casing that defines an inner duct wall having a fan rotor region;	a fan casing that defines an inner duct wall having a fan rotor region, **the inner duct wall of the fan casing at the fan rotor region being convergent**;
a hub disposed concentrically relative to the fan casing;	a hub disposed concentrically relative to the fan casing;
a fan rotor that includes multiple swept fan blades, the swept fan blades being spaced apart around the hub, each of the multiple swept fan blades having:	a fan rotor that includes multiple swept fan blades, the swept fan blades being spaced apart around the hub, each of the multiple swept fan blades having:
a tip profile that corresponds to the inner duct wall of the fan casing;	a tip profile that is **convergent so as to substantially** correspond to the **convergent** inner duct wall of the fan casing;
a leading edge that defines a variable sweep angle in a direction perpendicular to the axis of rotation, the leading edge including:	a leading edge that defines a variable sweep angle in a direction perpendicular to the axis of rotation, the leading edge including:
an inner region adjacent the hub, the inner region defining a forward sweep angle;	an inner region adjacent the hub, the inner region defining a forward sweep angle;
an intermediate region between the inner region and the fan casing, the intermediate region defining a rearward sweep angle; and	an intermediate region between the inner region and the fan casing, the intermediate region defining a rearward sweep angle; and
an outer region between the intermediate region and the fan casing, the outer region **being translated forward relative to a leading edge with the same sweep angle as an outward boundary of the intermediate region**.	an outer region between the intermediate region and the fan casing, the outer region **defining a forward sweep angle**.

22. The first feature in Rolls's claim missing from UTC's claim is that the fan stage is "at least in part" rotatable. This recitation adds nothing of substance to the Rolls claim. First of all, the recitation appears in a portion of the claim that simply introduces the structural features of the fan stage recited in the rest of the claim; that is, it does not actually recite any structural feature of the fan stage itself. In addition, the rotatable parts of the fan stage recited in UTC claim 18 (the hub and the rotor) are rotatable in the same fashion as the corresponding parts in Rolls claim 8.

23. The first feature with any substance in a technical sense that is included in Rolls's claim but is not found in UTC's claim is a duct wall that is convergent at the fan rotor region and a blade tip profile that is convergent so as to substantially correspond to the convergent inner duct wall of the fan casing.

24. One of ordinary skill in the art of designing fan stages for ducted fan gas turbine engines would have used a convergent duct wall and matching convergent blade tips for a number of reasons. First, fans with convergent casings and matching blade tip profiles had been conventional for decades by the time of the filing dates of the UTC '985 Patent and the Rolls '077 Patent, as shown in Fig. 1 of U.S. Patent 4,408,957 to Kurzrock et al. (attached as Exhibit 9), Fig. 2 of U.S. Patent 5,408,826 to Stewart et al. ("Stewart") (attached as Exhibit 10), Fig. 1 of Schwaar, and Fig. 1-14 of The Aircraft Gas Turbine Engine and Its Operation, Section 1, United Technologies Corporation, Part No. P&W 182408 (1988), page 1-17 (attached as Exhibit 12).

25. I note also that the Rolls '077 Patent does not assert that a convergent duct wall and matching blade tip profile in combination with the disclosed blade leading edge geometry provides any advantage that would not have been known to those skilled in the art. The Rolls '077 Patent, at column 6, lines 21-26, points to the discussion in the UTC '985 Patent of a convergent casing. However, in my opinion the Rolls '077 Patent misconstrues what the UTC

'985 Patent says in that regard. The discussion in question in the UTC '985 Patent actually says that the casing wall can be made convergent to ameliorate the problem of pressure wave reflection. At the same time, the '985 Patent teaches that mechanical and aerodynamic constraints may not permit convergence in a manner that eliminates the problems caused by shock waves discussed above in paragraphs 13-19. UTC '985 Patent, col. 3, lines 58-63. Then, just like the Rolls '077 Patent, the UTC '985 Patent says that the blade configuration provides the solution. UTC '985 Patent, col. 3, lines 64-67.

26. Furthermore, the prior art teaches the conventionality of providing UTC claim 18's fan with a convergent blade tip profile that substantially corresponds to a convergent fan casing. For example, Stewart, which discloses such structure, explains that it is important to make the radial clearance between the blade tip and the casing as small as possible. Stewart does this by providing an abradable coating 25 on the convergent casing and cutting a convergent path through the coating with the converging blade tips to ensure that the convergent casing and blade tips conform exactly. Stewart, col. 3, lines 45-50.

27. The only other difference between UTC claim 18 and Rolls claim 8 is that UTC claim 18 recites a blade leading edge outer region that is "translated forward relative to a leading edge with the same sweep angle as an outward boundary of the intermediate region." Rolls claim 8 recites this feature differently, calling for an outer region "defining a forward sweep angle."

28. As discussed in paragraph 19, the UTC '985 Patent teaches that the blade tip leading edge must be moved forward to intercept the shock wave adjacent the duct wall. Once one skilled in the art had been taught by UTC claim 18 to translate the outer region forward, that person would have inevitably provided a forward sweep angle for the reasons discussed above in paragraph 19. Stated another way, UTC claim 18 recites a blade that reduces leading edge sweep

in an outer or tip region for the same reason as the blade recited in Rolls claim 8: to move the shock behind the blade leading edge so the blade intercepts the shock near the fan casing. See Rolls '077 Patent, col. 4, lines 18-21. A fan engineer following that teaching would have been compelled to introduce the forward sweep recited in Rolls claim 8.

29. Therefore, it is my opinion that the subject matter of UTC claim 18 would have taught a fan engineer of ordinary skill the subject matter of Rolls claim 8.

30. UTC claim 18 does not explicitly call for intercepting the shock wave at the duct wall. However, UTC claim 23, which is otherwise identical to claim 18, recites that the blade tip leading edge sweep angle causes interception of that shock. Accordingly, UTC claim 23 merely explicitly recites a feature of Rolls claim 8's blade and of UTC claim 18's blade as taught by both the Rolls '077 Patent and the UTC '985 Patent. Thus, it is my opinion that the subject matter of each of UTC claim 18 and Rolls claim 8 would have taught a fan engineer of ordinary skill the subject matter of UTC claim 23.

31. UTC claim 23 does not include the limitations in Rolls claim 8 discussed in paragraphs 21-28. However, UTC claim 23 is identical to UTC claim 18, except that claim 23 explicitly recites the shock-intercepting feature of the blade of both UTC claim 18 and Rolls claim 8. Accordingly, the subject matter of UTC claim 23 would have taught a fan engineer of ordinary skill the subject matter of Rolls claim 8, for the reasons discussed in paragraphs 21-28.

32. The difference between Rolls '077 Patent claims 1 and 8 (other than differences in terminology that do not affect this analysis) is that claim 1's blade has a "stagger angle which increases progressively with blade height." However, fan engineers working at the time of the UTC and Rolls filing dates knew that to be a necessary feature of all fan blades.

33. Schwaar explains that a blade's stagger angle (or the "blade twist") must increase progressively with blade height because the blade's circumferential velocity ("V_x" in Fig. 2 of

Schwaar) progressively increases with blade height, while the axial airflow velocity ("V_y" in Fig. 2 of Schwaar) remains essentially constant. As Schwaar points out, it is a "basic consideration of blade design" that the twist angle "t" shown in Fig. 2 increases with blade height. Schwaar, col. 3, line 66, to col. 4, line 21. Accordingly, this limitation is not just obvious in view of the subject matter of Rolls claim 8; it is a necessary and inherent feature of any fan blade as explained in more detail in Exhibit 2, at paragraphs 20-21. Nor does providing a progressively increasing stagger angle perform any new function or achieve a different result when used in a blade that has a converging tip matching a converging duct wall and explicitly recited forward sweep in the tip region. That is, such a blade is subject to the same increasing velocity along the blade span as conventional blades, and must also have the increasing stagger angle feature in Rolls claim 1. Another way of saying the same thing is that by calling them "fan blades," Rolls claim 1 requires that they have a progressively increasing stagger angle. Otherwise, the blades could not perform the function required of them as part of a "fan stage for a ducted fan gas turbine engine."

Rolls Claims 2-7 And 9-13 Define The
Same Invention As UTC Claims 18 And 23

34. Rolls '077 Patent claim 2 depends from claim 1 and reads as follows:

> A fan stage of a ducted fan gas turbine engine as claimed in claim 1 wherein the blade has a tip region of about 20% of blade height characterised in that the stagger angle increases to approximately 70° at the tip relative to the airflow direction.

35. This claim recites a "tip region," but that term is entirely self-referential within claim 2 and has no other definition except that it constitutes an arbitrary amount (20%) of the blade's span. As a result, this limitation does not relate to any actual blade structure.

36. However, if this "tip region" is taken as being the portion of the blade in which the leading edge transitions from rearward sweep to claim 1's forward swept third height region, then one of ordinary skill in the art would have known to chose any suitable place on the leading

edge, such as 20% of the blade height for a particular ducted fan gas turbine engine, at which to begin the transition to forward sweep. Put another way, UTC claims 18 and 23, and Rolls claims 1 and 8, "teach" a blade with a leading edge profile with an outer region that is moved forward to provide certain aerodynamic advantages. A fan engineer designing for a particular engine, knowing the purpose of translating the blade outer region forward, would have found the proper place, such as the last 20% of the blade height, at which to begin the forward translation.

37. As for the recitation of a stagger angle at the blade tip of approximately 70° relative to the airflow direction, a fan engineer would have found this to be a routine matter of blade design for the same reasons that the recited extent of the forward swept tip region would have been obvious. That is, an ordinarily skilled fan engineer, faced with a set of engine operating conditions, would have simply made a blade with a stagger angle at the tip that would satisfy the requirement that the flow approach the blade at the proper angle. See Exhibit 2, paras. 20-21. That the angle might be 70° relative to the airflow direction for a given fan stage would simply be a matter of proper engineering. I note also that the stagger angle "t" at the blade tip as measured in Fig. 2 of Schwaar is about 70°, although I cannot be certain that drawing is to scale.

38. Dependent claim 3 of the Rolls '077 Patent reads as follows:

> A fan stage of a ducted fan gas turbine engine as claimed in claim 2 wherein [sic - in?] a blade tip region of about 20% of blade height the sweep of the leading edge changes from rearward sweep to forward sweep.

39. The subject matter of this claim would have been suggested to a fan engineer by UTC claims 18 and 23, and Rolls claims 1 and 8, for the same reasons discussed in paragraph 36 relating to claim 2.

40. Dependent claim 4 of the Rolls '077 Patent reads as follows:

> A fan stage of a ducted fan gas turbine engine as claimed in claim 3 wherein the blade is further characterised in that the stagger angle of the mid-height region of

> the blade is in the range from approximately 30° to approximately 55° relative to the airflow direction.

41. The subject matter of this claim would have been suggested to a fan engineer by the subject matter of UTC claims 18 and 23, and Rolls claims 1 and 8, for the same reasons discussed in paragraph 37 relating to claim 2. That is, the stagger angle of a fan blade at any particular point on the blade is simply a function of the engine operating conditions and the requirement that the flow approach the blade at the proper angle along the entire blade span. A fan engineer would have routinely designed a fan blade with the stagger angle characteristics in claim 4 if that is what the engine operating conditions dictated. I note also that Fig. 6 of Schwaar shows a swept fan blade with a stagger angle that measures between 30° and 55° in the blade mid-height region, although I cannot be certain that drawing is to scale.

42. Dependent claim 5 of the Rolls '077 Patent reads as follows:

> A fan stage of a ducted fan gas turbine engine as claimed in claim 1 wherein the sweep angle of the leading edge of a swept fan blade at a point on the leading edge is less than the complement of the angle of a Mach cone at any other point on the leading edge of the blade at greater radius from the root.

43. To understand why the feature in this claim is anticipated by UTC claims 18 and 23, and Rolls claims 1 and 8, it is first necessary to understand the concept of a "Mach cone." As explained in Exhibit 2, paragraph 13, describing aspects of supersonic flow using Mach cones and Mach angles is not new with the Rolls '077 Patent. Bliss is an example of an earlier reference that discusses Mach cone angles. Figs. 1C and 2 from Bliss, reproduced below, illustrate Mach angles and Mach cones associated with a blade leading edge:

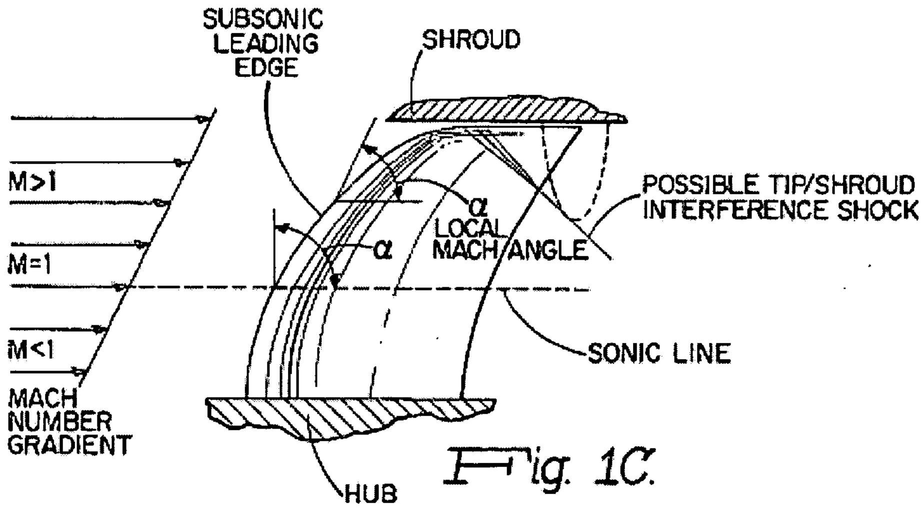


Fig. 1C.

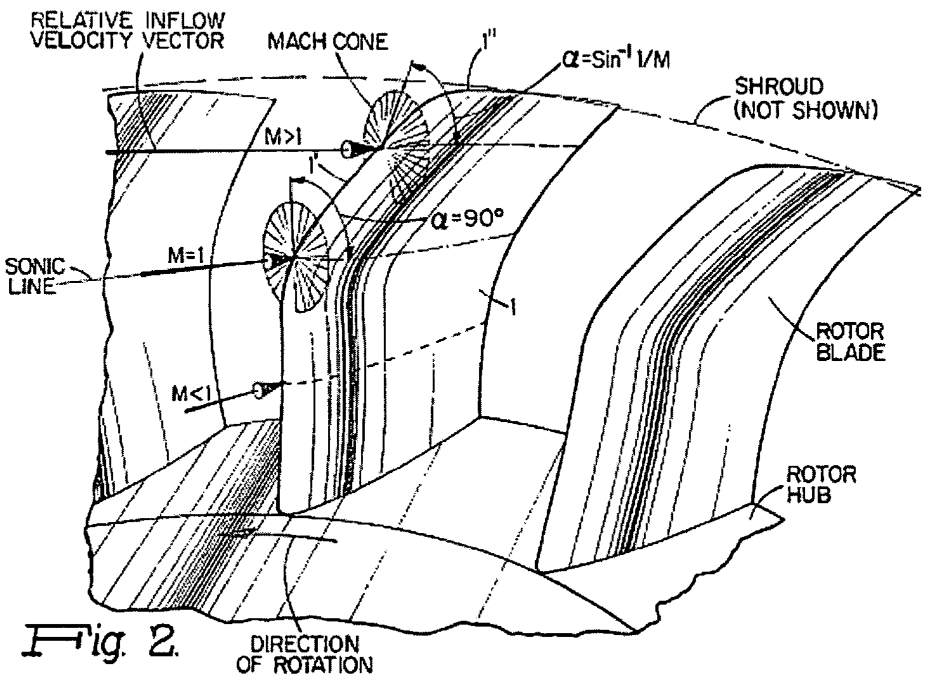


Fig. 2.

44. Bliss's Figure 1C shows the radial gradient associated with the velocity of the air approaching the blade, while Figure 2 is a three-dimensional depiction of theoretical Mach cones associated with points on a fan blade leading edge. (See Exhibit 2, Fig. 5 and para. 13.) Bliss shows a blade with a leading edge swept to a degree that it is always subject to subsonic velocities, thus theoretically eliminating the difficulties associated with shock waves. This is the opposite of Rolls claim 5, in that Bliss's sweep angle (call it "σ") is <u>greater</u> than the complement of the Mach cone angle α (that is, $\sigma > 90° - \alpha$).

45. In light of those known principles, Rolls claim 5 does no more than use different words to say the same thing as UTC claims 18 and 23 and Rolls claims 1 and 8. Put another way, consider what would happen if those UTC and Rolls claims did not meet claim 5: there would be no shock system to account for by using the leading edge configuration in those claims. Furthermore, UTC claim 23 explicitly recites the presence of a shock wave in the flow, meaning that its leading edge must have the Mach cone relationship recited in Rolls claim 5. Accordingly, the leading edge orientation in Rolls claim 5 is anticipated by UTC claims 18 and 23 and Rolls claims 1 and 8.

46. Dependent claim 6 of the Rolls '077 Patent reads as follows:

> A fan stage of a ducted fan gas turbine engine as claimed in claim 1 wherein the shape of the pressure surface of a swept fan blade and the suction surface thereof creates, in use, a line of minimum static pressure points on the suction surface of the blade, said line of minimum static pressure points is inclined with respect to the axial direction at a sweep angle which varies with span height of the blade, and has a negative value in a region of subsonic flow over the leading edge.

47. It will be appreciated immediately that, by definition, a gas turbine engine fan blade is an airfoil with a suction surface and a pressure surface (see Exhibit 2, Figure 6 and paras. 20-21; see also Fig. 3 of Weingold). Accordingly, at each location along the blade height, the suction surface will inherently have a minimum static pressure point. The locus of those points will, again by definition, be a line of minimum static pressure points.

48. As a practical matter, good fan blade design has always required that the line of minimum static pressure points for a swept fan blade essentially follow the leading edge profile, as shown in Weingold. In other words, given the leading edge geometry of the blades in UTC claims 18 and 23, and Rolls claims 1 and 8, and conventional fan design practice as exemplified by Weingold, an ordinarily skilled fan engineer would have inclined the lines of minimum static pressure points in those claims at a sweep angle that varied with blade height, and that sweep

angle would have been negative in the inner, subsonic-flow region of the blade, just as in Rolls claim 6.

49. Dependent claim 7 of the Rolls '077 Patent reads as follows:

> A fan stage of a ducted fan gas turbine engine as claimed in claim 6 wherein the sweep angle of the line of minimum static pressure points at a point on the line is less than the complement of a Mach cone angle at any other point on the line.

50. This claim relates to the same concept as claim 5. Mach cones associated with a blade's line of minimum static pressure points were known from Weingold, which is referred to in the Rolls '077 Patent at column 7, lines 25-27. Fig. 2a of Weingold illustrates Mach cones 38,44 associated with a line of minimum static pressure points:

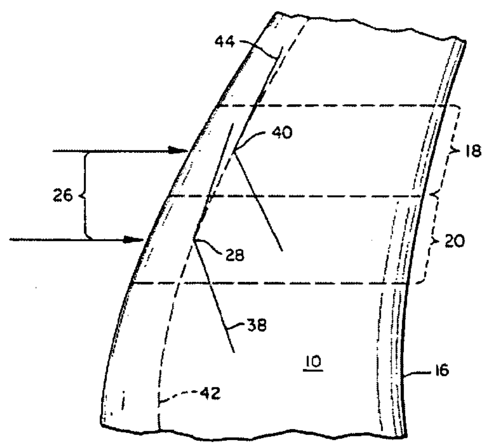


FIG. 2a

51. Weingold put the blade's maximum camber line 42 (associated with the line of minimum pressure points) behind the Mach cone associated with any inboard point of maximum camber. As in Bliss, the purpose of Weingold's geometry was to eliminate shock waves. Like Rolls claim 5, claim 7 puts the line of minimum static pressure points ahead of the Mach cone, the opposite of Weingold's geometry.

52. Accordingly, Rolls claim 7 does no more than recite a blade that creates shock waves, as discussed in paragraphs 44 and 45. But unless the blades in UTC claims 18 and 23, and Rolls claims 1 and 8, also create shock waves, there is no reason for them to have the

geometry that represents the invention embodied in those claims. And UTC claim 23 explicitly recites the presence of a shock system, which means it even more clearly anticipates claim 7's recitation of a particular orientation of the blade's line of minimum static pressure points.

53. Dependent claim 9 of the Rolls '077 Patent reads as follows:

> The fan stage according to claim 8, wherein the intermediate region extends further than the inner region along the axis of rotation.

54. This claim is ambiguous, since it is not clear what part of the blade's intermediate region is meant to "extend" further than the inner region along the rotational axis. Based on the disclosure of the Rolls '077 Patent at column 5, lines 23-51, and especially lines 26-28, I believe this claim means to say that the blade leading edge at the boundary between the inner and intermediate regions (segment S_5 in Figure 7a), is further upstream along the axis than the inner region leading edge (from the hub to segment S_5). As is the case with claims 5 and 7 (see paragraphs 42-52), this claim does no more than recite a feature of the blades in UTC claims 18 and 23, and Rolls claims 1 and 8. That is, a blade with forward sweep in the inner region and a rearward swept intermediate region must have claim 9's configuration. See Figures 1 and 2 of the UTC '985 Patent, Figures 3a, 5a, and 7a of the Rolls '077 Patent, and Figures 4 and 6 of Schwaar.

55. Dependent claim 10 of the Rolls '077 Patent reads as follows:

> The fan stage according to claim 8, wherein the inner duct wall of the fan casing at the fan rotor region is substantially convergent in the downstream direction.

56. This is already a feature of claim 8, so it adds nothing to distinguish it from claim 8's subject matter. It also recites a conventional feature of a ducted fan gas turbine engine, as discussed in paragraphs 23-26.

57. Dependent claim 11 of the Rolls '077 Patent reads as follows:

> The fan stage according to claim 8, wherein the tip profile of the swept fan blades are substantially convergent in the downstream direction.

58. As with claim 10, this is already a feature of claim 8, which means that it adds nothing to distinguish it from that claim. It also recites a conventional feature of a ducted fan gas turbine engine, as discussed above in paragraphs 23-26.

59. Dependent claim 12 of the Rolls '077 Patent reads as follows:

> The fan stage according to claim 8, wherein inner duct wall of the fan casing is not parallel to the tip profile of each of the multiple swept fan blades.

60. This claim actually contradicts claim 8, which recites that the blade tip profile is configured to "substantially correspond to the convergent inner duct wall." Nonetheless, it would have been well within the skill of a fan engineer also to provide a fan casing with an inner wall that is not parallel to the blade tip profile, as shown in U.S. Patent 4,012,165 to Kraig ("Kraig") (Fig. 1; movable door 32) (attached as Exhibit 11). That is, a fan engineer would have recognized that Kraig's door 32 would perform the same function in the fan stages recited in UTC claims 18 and 23, and Rolls claims 1 and 8, as it does in Kraig's fan.

61. Dependent claim 13 of the Rolls '077 Patent reads as follows:

> The fan stage according to claim 8, wherein each of the multiple swept fan blades includes a hub contacting surface that extends further than the tip profile along the axis of rotation.

62. This claim relates to physical properties of the blade rather than its aerodynamic performance. In my opinion the features recited in this claim are no more than the result of an ordinarily skilled fan engineer applying routine design techniques to determine optimum blade geometry within the performance and mechanical design parameters for the engine under consideration, and the Rolls '077 Patent does not say otherwise. That is, as discussed above with regard to other claims (see paragraphs 36, 37, and 41), a fan engineer would have provided a fan

blade with the geometry in claim 13 if performance and mechanical design criteria so dictated in making a blade in accordance with UTC claims 18 and 23 and Rolls claims 1 and 8.

UTC Claims 19 And 22 Define The Same Invention
As UTC Claims 18 And 23 And Rolls Claims 1 And 8

63. Dependent UTC claim 19, set forth in paragraph 67, is based closely on dependent claim 9 of the Rolls '077 Patent. Its subject matter is in accord with my understanding of the subject matter intended to be covered by Rolls claim 9, as discussed in paragraph 54. UTC dependent claim 22 is identical to Rolls claim 13.

64. Accordingly, I believe that a fan engineer would have concluded that the subject matter of each of UTC claims 19 and 22 have the same relation as each of Rolls claims 9 and 13 to the subject matter of UTC claims 18 and 23, and Rolls claims 1 and 8, for the reasons discussed above in paragraphs 54 and 62.

The '985 Patent Specification Discloses The
Subject Matter Of UTC Claims 18, 19, 22, And 23

65. The following discussion applies the terms of these UTC claims to the disclosure in the UTC '985 Patent.

Independent Claim 18

66. The following chart applies the terms of claim 18 to the disclosure in the UTC '985 Patent.

Present Application Claim 18	UTC '985 Patent Disclosure
A fan stage of a ducted fan gas turbine engine that is rotatable about an axis of rotation and defines a downstream direction along the axis of rotation, comprising:	A gas turbine engine fan stage 10 has blades 12 circumscribed by a case 42 that forms a fan duct (Figs. 1 and 2; col. 2, lines 42-44 and 56-58). The fan is rotatable about an axis 18 (Fig. 2, col. 2, lines 44-46).
a fan casing that defines an inner duct wall having a fan rotor region;	The case 42 defines an inner wall in a region that is axially coextensive with the blade 12 (Figs. 1 and 2; col. 3, lines 58-67).

a hub disposed concentrically relative to the fan casing;	The hub 16, 20 and case 42 have the same centerline 18 and thus are concentric (Figs. 1 and 2; col. 2, lines 42-63).
a fan rotor that includes multiple swept fan blades, the swept fan blades being spaced apart around the hub, each of the multiple swept fan blades having:	The fan blades 12 are spaced apart around the hub. The blades are swept (col. 4, lines 11-17 and 24-29).
a tip profile that corresponds to the inner duct wall of the fan casing;	The blade 12, seen in Fig. 2 projected into the radial plane of the drawing, has a tip profile that corresponds to the inner wall of the case 42 (Figs. 1 and 2; col. 3, lines 26-29).
a leading edge that defines a variable sweep angle in a direction perpendicular to the axis of rotation, the leading edge including:	The blade leading edge 28 has an intermediate region 70 with a sweep angle σ_1 and a tip region 74 with a sweep angle σ_2, both of which vary with increasing blade height (Fig. 2; col. 4, lines 11-17 and lines 24-29).
an inner region adjacent the hub, the inner region defining a forward sweep angle;	A first height region between the blade root 24 (that is, adjacent the hub 16,20) and $r_{t\text{-inner}}$ is swept forward (Fig. 2; col. 5, lines 55-58).
an intermediate region between the inner region and the fan casing, the intermediate region defining a rearward sweep angle; and	The intermediate region 70 between $r_{t\text{-inner}}$ and $r_{t\text{-outer}}$ is swept rearward (Figs. 1 and 2; col. 4, lines 11-13).
an outer region between the intermediate region and the fan casing, the outer region being translated forward relative to a leading edge with the same sweep angle as an outward boundary of the intermediate region.	The tip region 74 between $r_{t\text{-outer}}$ and the blade tip 26 is translated axially forward relative to a conventional blade (col. 4, line 62, to col. 5, line 5).

<u>Dependent Claims 19 And 22</u>

67. Dependent claim 19 conforms to my understanding of the intended meaning of Rolls claim 9, discussed in paragraph 54. Claim 19 reads as follows:

> The fan stage according to claim 18, wherein the leading edge at a boundary between the intermediate region and the inner region extends further upstream along the axis of rotation than the inner region.

68. Fig. 2 of the UTC '985 Patent shows the subject matter of this claim. That is, Fig. 2, being a projection of the blade onto the plane of the figure (see col. 3, lines 26-29), shows the blade at $r_{t\text{-inner}}$ extending further upstream along the axis 18 than the region inward thereof.

69. Dependent claim 22 is identical to dependent claim 13 of the Rolls '077 Patent. It reads as follows:

> The fan stage according to claim 18, wherein each of the multiple swept fan blades includes a hub contacting surface that extends further than the tip profile along the axis of rotation.

70. Fig. 2 of the UTC '985 Patent shows that the blade's hub contacting surface at the root 24 extends further along the axis 18 than at the blade tip 26.

Independent Claim 23

71. The following chart applies the terms of claim 23 to the disclosure in the UTC '985 Patent.

Present Application Claim 23	UTC '985 Patent Disclosure
A fan stage of a ducted fan gas turbine engine that is rotatable about an axis of rotation and defines a downstream direction along the axis of rotation, comprising:	A gas turbine engine fan stage 10 has blades 12 circumscribed by a case 42 that forms a fan duct (Figs. 1 and 2; col. 2, lines 42-44 and 56-58). The fan is rotatable about an axis 18 (Fig. 2, col. 2, lines 44-46).
a fan casing that defines an inner duct wall having a fan rotor region;	The case 42 defines an inner wall in a region that is axially coextensive with the blade 12 (Figs. 1 and 2; col. 3, lines 58-67).
a hub disposed concentrically relative to the fan casing;	The hub 16, 20 and case 42 have the same centerline 18 and thus are concentric (Figs. 1 and 2; col. 2, lines 42-63).
a fan rotor that includes multiple swept fan blades, the swept fan blades being spaced apart around the hub and being capable of rotating at speeds providing supersonic working medium gas velocities over the blades to cause a shock in the gas adjacent the inner duct wall, each of the multiple swept fan blades having:	The fan blades 12 are spaced apart around the hub. The blades are swept (col. 4, lines 11-17 and 24-29). The blades rotate at speeds high enough to experience supersonic velocities near their tips, and a shock 64 forms adjacent the inner duct wall (Fig. 2; col. 3, lines 30-40).

a tip profile that corresponds to the inner duct wall of the fan casing;	The blade 12, seen in Fig. 2 projected into the radial plane of the drawing, has a tip profile that corresponds to the inner wall of the case 42 (Figs. 1 and 2; col. 3, lines 26-29).
a leading edge that defines a variable sweep angle in a direction perpendicular to the axis of rotation, the leading edge including:	The blade leading edge 28 has an intermediate region 70 with a sweep angle σ_1 and a tip region 74 with a sweep angle σ_2, both of which vary with increasing blade height (Fig. 2; col. 4, lines 11-17 and lines 24-29).
an inner region adjacent the hub, the inner region defining a forward sweep angle;	A first height region between the blade root 24 (that is, adjacent the hub 16,20) and $r_{t\text{-inner}}$ is swept forward (Fig. 2; col. 5, lines 55-58).
an intermediate region between the inner region and the fan casing, the intermediate region defining a rearward sweep angle; and	The intermediate region 70 between $r_{t\text{-inner}}$ and $r_{t\text{-outer}}$ is swept rearward (Figs. 1 and 2; col. 4, lines 11-13).
an outer region between the intermediate region and the fan casing, the outer region being translated forward relative to a leading edge with the same sweep angle as an outward boundary of the intermediate region to provide a sweep angle that causes the blade to intercept the shock.	The tip region 74 between $r_{t\text{-outer}}$ and the blade tip 26 is translated axially forward relative to a conventional blade to intercept the shock (col. 4, line 62, to col. 5, line 5).

I hereby declare that all statements made herein of my own knowledge are true and that all statements made on information and belief are believed to be true; and further that the statements were made with the knowledge that willful false statements and the like so made are punishable by fine or imprisonment, or both, under § 1001 of Title XVIII of United States Code, and that such willful false statements made jeopardize the validity of this application or any patent issued thereon.

Date: 16th May 2003

F. Breugelmans

Frans A.E. Breugelmans

Exhibit 1

Curriculum Vitae - Prof. F. A. E. Breugelmans

Von Karman Institute for Fluid Dynamics
72 Steenweg op Waterloo
1640 Sint Genesius Rode
Belgium

Born: Diest, Belgium, December 4, 1938
Nationality: Belgian

Professional Experience:

1962	Mechanical Engineering, K.U. Leuven, Civil Engineer (MS)
1963	Postgraduate TCEA (now VKI), specialty in turbomachinery
1965	Assistant VKI, turbomachinery laboratory
1967-1996	Lecturer, K.U. Leuven (course: Turbomachines)
1977	Full professor, VKI
1977-1997	Head of Turbomachinery Department, VKI
1977-1997	NATO, AGARD Propulsion & Energetics Panel (PEP) - panel member
1975-pres.	International Society for Airbreathing Propulsion, ISOABE, International administrative secretary
1976-1977	Iowa State Univ., visiting assoc. professor, Mech. Engrg. Dept.
1991-2000	Assistant director, VKI
1994	Two months as visiting scientist in Aeronautics Dept. Tokyo Univ. by invitation of Japanese Program for Promotion of Science
1995-pres.	International Committee member, Japanese Gasturbine Conference
1996-pres.	Associate lecturer, K.U. Leuven (course: Turbomachines for 5th year Mechanical Engrg.)
1997	NATO, RTO-AVT (Research & Technology Organization, Advanced Vehicle Technology, AGARD PEP) - panel member
1998-pres.	NATO, RTB (Research & Technology Board) - Belgian national delegate (one of three delegates)
Feb. 2000	Honorary professor, VKI (with teaching & consulting activities) (retirement date)
June 2000	Invitation by IADF (International Aircraft Development Fund) of Japanese International Exchange Program, June 8-24, 2000, in Japan
May 2002	Gas Turbine Society of Japan, 30th anniversary ceremony, invited guest

Memberships:

KVIV (Royal Flemish Engrg. Society) (40 years)
KVIV Working group for Aerospace (20 years)
International Society for Airbreathing Engines (31 years)
Belgian Association of Mech. Engineers (30 years)
American Society of Mechanical Engineering (28 years)

USAF Windows on Science program, seminars given at Wright-Patterson AFB (invitations in 1988 and 1998)
Program Chairman, AGARD PEP 90 Conference in Brussels 1997
Local coordinator ASME Gasturbine Conference in Brussels 1990
International Committee Member, Japanese Society for Gas Turbines
Security clearance: NATO secret (as RTB member)

Research:

Supersonic compressor design, construction and testing (USAF)	1962-1977
Supersonic compressors and cascades (USAF, Rolls-Royce)	1962-1977
Consulting on compressor for 20 MW gasturbine (KWU)	1969
Consulting on transonic multistage compressors for industrial applications (2 projects) (Nuovo Pignone, Westinghouse)	1970-1976
Supersonic cascades, development of blade sections for transonic and supersonic compressors and fans (Aerospace Research Lab.-WPAFB, Rolls-Royce, Caterpillar)	1962-1977
Consulting and testing on single-stage transonic compressors for industrial applications (6 models) (Rateau)	1965-1975
Consulting and testing on two-stage transonic compressors for industrial applications (2 models) (Rateau)	1970-1975
Gasturbine design and analysis (FN, Thomassen, Heron, et al.)	1975-1985
Centripetal turbine expanders (ACEC, Borsig, Cockerill, et al.)	1975-2000
Steamturbine CAD development (ACEC, Thermodyn)	1987-1998
Riblets, sweep, dihedral in compressors (Toshiba, VKI)	1981-1999
Compressor stall (VKI)	1975-1999
Chaos theory application (VKI, EEC Proposal)	1991-2000
Space propulsion with LOX collection (Techspace)	1997-2000
Future large space transport systems (proposal)	2000-2003
Optical methods for turbomachines (PIV, Holography)	1994-1999

Patent:

U.S. Publ. No. US2002/0066267, "Process and Device for Collecting Air, Engine Associated Therewith," June 6, 2002 (with TechSpace Aero Belgium)

Teaching:

VKI lectures in:
Introduction, Cycles - Propulsion
Radial equilibrium in turbomachines
Advanced axial compressors
Design exercise of axial flow compressor

Organization of VKI Lecture Series:
High speed propulsion (rockets)
High speed propulsion (ramjets and scramjets)
Various lecture series on axial flow compressors
Various lecture series on advanced measurement techniques
Special course on turbomachinery, K.U. Leuven
Advanced axial compressor course at VKI, as Honorary Professor (Feb. 2000)

Publications in:

ASME - IGTI conferences and journals
ISOABE symposia
AGARD conference proceedings
RTO - AVT conference proceedings
VKI publications
JSME (Japanese Society of Mechanical Engineers) conference
GTSJ (Gas Turbine Society of Japan) conferences
ISUUAT conference
SFME (Societe Français des Mécaniciens)
(List of publications attached)

Coordinates:

Home Address:
Prof. F. Breugelmans
52 Schoonaerde
B-3290 DIEST
BELGIUM

Business Address:
Prof. F.Breugelmans
von Karman Institute for Fluid Dynamics
72 Steenweg op Waterloo
1640 Sint Genesius Rode
Belgium
Tel: 32-(0)2-359-9611
Fax: 32-(0)2-359-9600

PUBLICATIONS:

1. A Freon Regenerating System for the R-2 Compressor Test Facility at TCEA, VKI, IN 2, July 1963.

2. Design Data of the R-32 Rotor (a Supersonic Axial Compressor Using Blunt Trailing Edge Blades), VKI, IN 3, Oct. 1963.

3. Description of VKIFD R-2 High Speed Compressor Facility, VKI IN 5, Feb. 1964 (with J. Chauvin, F. Thiry).

4. Description and Calibration of the Pressure and Temperature Probes for the R-2 Rotor Test Rig, VKI IN 7, May 1964.

5. ISM 1620 Calculations and Data Reduction Programs for the VKIFD Turbomachinery Laboratory, VKI IN 4, May 1964 (with J. Chauvin and J. Laurensis).

6. Preliminary Investigation of a Blunt Trailing Edge Blade Cascade in the S-3 Supersonic Cascade Tunnel, VKI IN 8, Jan. 1964.

7. Investigation of a one-Stage Axial Flow Supersonic Compressor. Part I, (part II, with Kiock), VKI IN 12, March 1965.

8. On the Use of a Pseudo-Shock System in Supersonic Compressors, Paper at WGLR Meeting, München, 1965.

9. The Blunt Trailing Edge Axial Flow Supersonic Compressor, Aerodynamics of Powerplant Installation. AGARDograph 103, 1965.

10. Cascade Data for High Camber Blunt Trailing Edge Blades,VKI IN 17, Nov. 1966 (with C. Sleverding).

11. Rotor Performance of the R-32-II Blunt Trailing Edge Blade Supersonic Compressor, VKI IN 20, April 1967.

12. Note on the Concept of Blunt Trailing Edge Blades, VKI IN 21, Jan. 1967.

13. Application of Blunt Trailing Edge Bladings to Supersonic Compressor Stages, VKI IN 22, May 1967 (with J. Chauvin).

14. Supersonic Compressors, VKI CN 67, April 1967, also AGARDograph 120.

15. High Speed Cascade Testing and its Application to Axial Flow Supersonic Compressors, ASME Paper 68-GT-10, March 1968.

16. Perspectives and Problems of Transonic and Supersonic Compressors, AGARD Conference Proceedings 34, Sept. 1968.

17. Application of the Blunt Trailing Edge Blade Concept to Low Hub-Tip Ratio Transonic-Supersonic Compressors, VKI IN 27, April 1968 (with Chauvin and Griepentrog), VKI IN 28, April 1969 (with Chauvin, Griepentrog and Rottlers).

18. Blunt Trailing Edge Blading Studies in Low and High Speed Flows (co-author), VKI IN 35, March 1970.

19. A Review of Transonic and Supersonic Compressor Research at the von Karman Institute, paper presented at Seminar on Axial Flow Compressors, The Hatfield Polytechnic, October 28, 1970.

20. High Speed Compressor Fundamentals, UTSI Lecture, Nov. 30 - Dec. 4, 1970.

21. Problems and Prospects on Transonic and Supersonic Compressors, UTSI Lecture Short Course on "Advanced Air-Breathing Propulsion Technology and Systems", Nov. 30 - Dec. 4, 1970 (as AGARD consultant).

22. Survey of Probes, Pressure, Temperature and Flow Direction Measurements, co-author G. Junkhan (Iowa State Univ.), VKI CN 82, Oct. 1971/Jan. 1973.

23. Flow in Turbomachines and Turbomachinery Component Characteristics, VKI CN 83, Nov. 1971.

24. The Supersonic Compressor Stage, VKI Short Course - LS 47, VKI CN 86, April 1972.

25. The Supersonic Compressor Research at the von Karman Institute for Fluid Dynamics. 1st International Symposium on Air Breathing Engines; Marseille, June 19-23, 1972.

26. The Low Hub-Tip Ratio Supersonic Axial Flow Compressor, Original model - Volume I, VKI IN 40, June 1972.

27. The Low Hub-Tip Ratio Supersonic Axial Flow Compressor. Inlet Blockage - Volume II, VKI IN 41, Nov. 1972.

28. The Low Hub-Tip Ratio Supersonic Axial Flow, Compressor Bleed - Volume III, VKI IN 42, Jan. 1973.

29. The Low Hub-Tip Ratio Supersonic Axial Flow Compressor, Rotor Performance - Volume IV, VKI IN 46, Aug. 1973.

30. Transonic Flows in Turbomachinery, VKI LS 59, May 1973.

31. The cascade and rotor section performance of a $9^{0}57$ cambered DCA Airfoil - Comparison with 2D Calculations in the Transonic and Supersonic Flow Regimes, VKI LS 59, May 1973.

32. The cascade and static section performance of a 48^0 cambered DCA Airfoil - Comparison with 2D Flow Calculations in the Compressible Flow Regime, VKI LS 59, May 1973.

33. The Mach 2 Axial Flow Compressor Stage, Houston, TX, ASME 75-GT-22, June 1975.

34. Investigation of the axial velocity density ratio in a high turning cascade, DFVLR, ASME 75-GT-26, March 1975 (with H. Starken and P. Schimming).

35. Introduction to Measurement Techniques in Turbomachinery, VKI LS 78, April 1975.

36. Investigation of the Axial Velocity Density Ratio in a High Turning Cascade, Houston, TX, ASME 75-GT-25, June 1975.

37. The Supersonic Axial Inlet Component in a Compressor, Houston, TX, ASME 75-GT-26, June 1975.

38. Advanced Testing Techniques in Turbomachines, VKI LS 78, April 1975.

39. The Effect of Leading Edge Thickness on the Bow Shock in Transonic Rotors, Squid Workshop – USNPGS, Monterey, CA, Feb. 1976, published in "Transonic Flow Problems in Turbomachinery," Hemisphere Publ. Corp.

40. Supersonic Compressors with Subsonic and Supersonic Axial Inlet Component, Paper 76-017, 3rd International Symposium on Air Breathing Engines, München, March 1976.

41. Industrial compressors, VKI LS 91, May 1976.

42. The Reynolds Number Effect in Axial Flow Compressors, VKI LS 91, May 1976.

43. The Erosion Effect in Axial Flow Compressors, VKI LS 91, May 1976.

44. The Clearance Effects in Axial Flow Compressors, VKI LS 91, May 1976.

45. The Casing Treatment in Axial Flow Compressors, VKI LS 91, May 1976.

46. Variable Geometry Applied to Supersonic Compressors, Philadelphia, PA, ASME 77-GT-35, June 1977.

47. Iowa State Univ. Lecture Notes on Flow in Turbomachines, 1976-77 academic year.

48. Unsteady Flow in Turbomachines, VKI LS 1979-3, Jan. 1979.

49. Measurements of the Flow Field in a Rotating Stall Cell, VKI LS 1979-3, Jan. 1979.

50. Blade-wake Interactions and Temperature Effects, VKI LS 1979-3, Jan.1979.

51. Reverse Flow, Pre-rotation and Unsteady Inlet Flow in Centrifugal Pumps, N.E.L., Nov. 1979.

52. A Hot Wire Probe for Measurement of Flow Near the Surface of a Compressor Blade, Euromech Coll. 132, July 1980.

53. Turbine Blade Cooling Research at VKI, 5th International Symposium on Air Breathing Engines Bangalore, Feb. 1981.

54. Gas Turbine Related Research at VKI. Japanese Society of Gasturbine Engineers, Tokyo, Sept. 1981 (seminar).

55. The VKIFD Turbomachinery Department - Facilities & Experimental Research, Revue <M>, Vol. 28, No 1, March 1982, pp. 17-21.

56. Aerodynamic Research in Turbomachines at VKI - Gas Turbine Related Problems, Part I: Axial Flow Compressors, Journal of Gas Turbine Society of Japan, Vol. 10, No. 39, Dec. 1982, pp. 11-22.

57. A Sensor for Flow Measurement near the Surface of a Compressor Blade, P.M. Ligrani, R.R. Gyles, K. Mathioudakis, F.A.E. Breugelmans, Journal of Physics E: Scientific Instruments, Vol. 16, No. 5, May 1983, pp. 431-437.

58. Influence of Dihedral on the Secondary Flow in a 2-D Compressor Cascade, F.A.E. Breugelmans, Y. Carels, M. Demuth, ASME Transactions, Series A: Journal Engineering for Gas Turbines & Power, Vol. 106, No. 3, July 1984, pp. 578-584.

59. Rotating Stall Cells in a Low Speed Axial Flow Compressor, F.A.E. Breugelmans, K. Mathioudakis, F. Casalini, AIAA, Journal of Aircraft, Vol. 22, No. 3, March 1985, pp. 175-181.

60. Use of Triple Hot Wires to Measure Unsteady Flows with Large Direction Changes, K. Mathioudakis, F.A.E. Breugelmans, Journal of Physics: Scientific Instruments, Vol. 18, No. 5, May 1985, pp. 414-419.

61. Prerotation and Fluid Recirculation in the Suction Pipe of Centrifugal Pumps, F.A.E. Breugelmans, M. Sen, Revue <M> and 11th Turbomachinery Symposium 1982, Houston, TX, Dec. 1982.

62. Flow in Rotating Stall Cells of a Low Speed Axial Flow Compressor, F.A.E. Breugelmans, K. Mathioudakis, F. Casalini, 6th International Symposium on Airbreathing Engines. Paris, June 1983, pp. 632.

63. Measurement of the Radial Flow along a Low Speed Compressor Blading During Unstalled and Stalled Operation, Paper 83 IGT-79, F.A.E. Breugelmans, L. Lambropoulos, K. Mathioudakis, Tokyo International Gas Turbine Congress, Oct. 1983.

64. Structure of Rotating Stall Cells: 1. Absolute Motion, 2. Relative Motion, K. Mathioudakis, F.A.E. Breugelmans, Unsteady Flow in Turbomachines, VKI LS 1984-2, Feb. 1984.

65. Hot Wire Anemometry in Turbomachinery Components, K. Mathioudakis, F.A.E. Breugelmans, Measurement Techniques in Turbomachinery, VKI LS 1985-3, March 1985.

66. Development of Small Rotating Stall in Axial Compressors, ASME paper 85-GT-227, K. Mathioudakis, F.A.E. Breugelmans, ASME Gas Turbine Conference, Houston, TX, March 1985.

Publications within last 15 years:

1. Experimental Convective Heat Transfer Investigation Around a Film Cooled High Pressure Turbine Blade, C. Camci, T. Arts, F.A.E. Breugelmans, AIAA-ISABE 85-7035, 7th International Symposium on Air Breathing Engines, Beijing, Sept. 1985.

2. Influence of Incidence Angle on the Secondary Flow in a Compressor Cascade with Different Dihedral Distribution, F.A.E. Breugelmans, AIAA-ISABE 85-7078, 7th International Symposium on Air Breathing Engines, Beijing, Sept. 1985.

3. The Three Dimensional Flow Field Inside an Axial Compressor Operating in Rotating Stall, K. Mathioudakis, F.A.E. Breugelmans, AIAA-ISABE 85-7086, 7th International Symposium on Air Breathing Engines, Beijing, Sept. 1985.

4. Unsteady Loss in a Low Speed Axial Flow Compressor During Rotating Stall, F.A.E. Breugelmans, L. Huang, L. Larosiliere, P. Andrew, AIAA-ISABE89-7026, 9th International Symposium on Air Breathing Engines, Athens, Sept. 1989.

5. Three Dimensional Flow Measurements in the Relative Frame of Reference During Rotating Stall, F.A.E. Breugelmans, E.M. Blanco, AIAA-ISABE91-7100, 10th International Symposium on Air Breathing Engines, Nottingham, Sept. 1991

6. Unsteady Flow in Axial Flow Compressor, Modern Research Topics in Aerospace Propulsion (with G. Angelino, L. De Luca, W.A. Sirignano), pp. 275-295, Springer Verlag 1991

7. Rotating Stall-Modelling of Rotating Stall Cells, F.A.E. Breugelmans, P. Lippert, VKI LS 1992-2, Jan. 1992.

8. Rotating Stall-Measurement Techniques, F.A.E. Breugelmans, L. Huang, E. Blanco, VKI LS 1992-2, Jan. 1992.

9. Rotating Stall-Relative Frame, Unsteady Loss, F.A.E. Breugelmans, L. Larosiliere, L. Huang, Ph. Andrew,VKI LS 1992-2, Jan. 1992.

10. Rotating Stall-Relative Frame, Unsteady Flow Field, F.A.E. Breugelmans, E. Blanco, VKI LS 1992-2, Jan. 1992.

11. Rotating Stall-Leading Edge Vortex, Energy Dissipation and Chaos Theory Application, F.A.E. Breugelmans, P. Lippert, VKI LS 1992-2, Jan. 1992.

12. The Leading Edge Vortex of a Rotating Stall Cell, F.A.E. Breugelmans, E. Fabry, J. Mijs, IAA-ISABE 93-7009, 11th International Symposium on Air Breathing Engines, Tokyo, Sept. 1993.

13. The Energy Dissipation in a Rotating Stall Cell, O. Eertink, F.A.E. Breugelmans, AIAA-ISABE 93-7010, 11th International Symposium on Air Breathing Engines, Tokyo, Sept. 1993.

14. Application of Strange Attractors to the Problem of Rotating Stall, 7th ISUAAT, Fukuoka 1994, pp 585 - 597 in "Unsteady Aerodynamics and Aeroelasticity of Turbomachines," Elsevier 1995

15. Study of the Blade-to-Blade Flow in a Compressor Rotor using PIV, D. Tisserant, F.A.E. Breugelmans, AIAA-ISABE 95-7060, 12th International Symposium on Air Breathing Engines, Melbourne, Sept. 1995

16. Phase Portraits from Rotating Stall Time Series, C. Palomba, F.A.E. Breugelmans, AIAA-ISABE 95-7063, 12th International Symposium on Air Breathing Engines Melbourne, Sept. 1995.

17. Combined Cycle for SSTO Rocket: Definition of the Key Technologies, V. Balepin, F.A.E. Breugelmans, AIAA-ISABE 97-7132, 13th International Symposium on Air Breathing Engines, Chattanooga, TN, Sept. 1997.

18. Parameter Study of the Mixed Turbofan Power Unit for an In-Flight LOX Collection System, F.A.E. Breugelmans, R. Van Raebroeck, AIAA-ISABE 97-7171, 13th International Symposium on Air Breathing Engines, Chattanooga, TN, Sept. 1997.

19. Rotor Blade-to-Blade Measurements Using Particle Image Velocimetry, D. Tisserant, F.A.E. Breugelmans, Transactions of the ASME, Journal of Turbomachinery, Vol. 119, April 1997, p. 176.

20. Strange Attractor Characterisation of Rotating Stall in an Axial Flow Compressor, C. Palomba, L. van der Horst, F.A.E. Breugelmans, JSME, International Conference on Fluid Engineering, Tokyo, July 1997

21. Recent Progress in Whole-Field Non-Intrusive Measurements of 3-D Velocity Vectors and Particle Size, C.S. Moraitis, D. Tisserant, F.A.E. Breugelmans, JSME, International Conference on Fluid Engineering, Tokyo, July 1997.

22. Energy Efficient Design through Fluid Dynamics Improvements in Turbocomponents, International Seminar on Thermal and Fluid Engineering for Advanced Energy Systems, Kyushu University, Kasuga, Japan, July 23-24, 1997.

23. Comparison of Sweep and Dihedral Effects on Compressor Cascade Performance, T. Sasaki, F.A.E. Breugelmans, Transactions ASME, Journal of Turbomachinery, Vol. 120, July 1998, p. 454.

24. Récherche Récente en Décrochage Tournant, F.A.E. Breugelmans, C. Palomba, Revue Française de Mécanique No. 1998-4, Décembre 1998, p. 277.

25. Experimental Investigation of Sweep and Dihedral in Compressors, Lecture in VKI-LS "Turbomachinery Blade Design Systems," 1999-2, Feb. 1999.

26. Design and Experimental Verification of an Optimized Compressor Blade, P.F. Bogers, R.A. Van den Braembussche, F.A.E. Breugelmans, ASME 98-GT-193, Stockholm 1998.

27. Investigation of Riblets in a CDB, DCA and 65-S Compressor Cascade, S. Nagao, F.A.E. Breugelmans, Proceedings of the International Gas Turbine Congress 1999, Kobe, Nov. 14-19, 1999, p. 445.

28. Fully and Semi-Reusable T.S.T.O Using In-Flight LOX Collection, P. Hendrick; B. Marquet, M. Strengart, F.A.E. Breugelmans, Proceedings 3rd European Aerospace Conference, Paris, Dec. 1999.

29. Perspectives of Phase Portraits in the Detection of Compressor Instabilities, M. D'Ishia, F.A.E. Breugelmans, NATO-RTO Symposium on Active Control Technology, Braunschweig, May 2000.

30. Experimental Investigation of the Blade-to-Blade Flow in a Compressor Rotor by Digital Particle Image Velocimetry, N. Balzani, F. Scarano, M.L .Riethmuller, F.A.E. Breugelmans, ASME-IGTI 2000-GT-55 Gas Turbine Conference, München, May 2000, Transactions ASME, Journal of Turbomachinery, Vol. 122. No. 4, Oct. 2000, pp. 743-751.

31. Energy Efficient Design through Fluid Dynamics Improvements in Turbocompressors, "Some Aero-Thermo-Fluid Aspects in Airbreathing Propulsion," ed. CIAM Moscow & University of Tokyo, ISBN 5-94049-004-2, 2001, pp. 44-58.

32. Hypersonic Propulsion, TI-KVIV Engineering Association, evening lecture, 105 pages of transparencies, VKI seminars 2002/2003.

33. Propulsion for Launch and Interplanetary Travel, Euroavia-VKI Scientific and Educational Symposium: Mission to Mars, Nov. 2002.

34. Numerous contract reports.

VKI Lecture Series:

1. Industrial Compressors. May 1976.

2. Combustion Problems in Gas Turbine Applications. January 1977.

3. Closed Cycle Gas Turbines. May 1977.

4. Off-Design Performance of Pumps. March 1978.

5. Pumps for Nuclear Applications. March 1978.

6. Unsteady Flow in Turbomachines. January 1979.

7. Separated Flows in Turbomachinery Components. January 1981.

8. Measurement Techniques in Turbomachines. May 1981.

9. Unsteady Flows in Turbomachines. February 1984.

10. Transonic Compressors. February 1988.
    a. Variable Geometry in Supersonic Compressors (contributed 21 pages).
    b. Axial Supersonic Inlet Component (contributed 26 pages).

11. Measurement Techniques in Aerodynamics. April 1989.

12. Axial flow Compressors. January 1992.
    Rotating Stall (contributed 195 pages in five sections).

13. Spacecraft Propulsion. January 1993.

14. Measurement Techniques. April 1993.

15. Non-Newtonian Mechanics. February 1994 (Local coordinator).

16. Measurement Techniques. January 1995.

17. Unsteady Flow in Turbomachines. March 1996.

18. High Speed Propulsion. January 1998.

19. Advanced Measurement Techniques. April 1998.

20. Turbomachinery Blade Design Systems. February 1999.
    Experimental Investigation of Sweep and Dihedral in Compressors (contributed 61 pages).

21. Noise in Turbomachines. February 2000.

(2)

TECHNOLOGY FUNDAMENTALS

1. The following schematic illustrates a typical jet engine (or more properly, a turbojet or gas turbine engine):

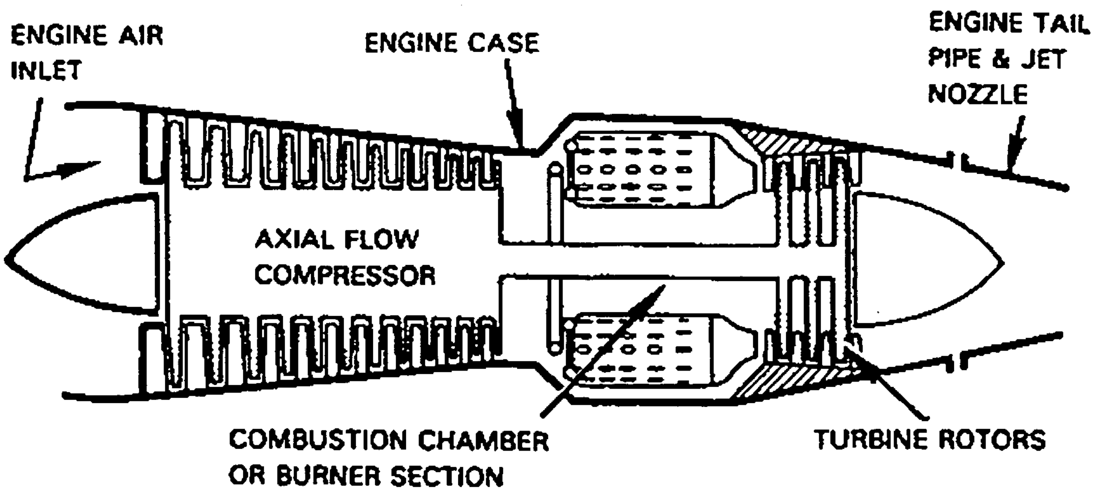


FIGURE 1[1] - Basic Turbojet Engine

2. A turbojet engine creates thrust by taking in air at an engine air inlet at one velocity and expelling it from a jet nozzle at a higher velocity. The engine is enclosed within an engine case, and air entering the inlet is compressed in an axial flow compressor. Air from the compressor enters a combustion chamber, fuel is added to the compressed air, and the fuel-air mixture is burned in the combustion chamber. The combustion products pass through a turbine with rotors that use some of the energy generated by burning the compressed air to drive the compressor through a shaft connecting the compressor and turbine rotors. The combustion products exiting the turbine into the engine tail pipe still have a great deal of energy (in the form of increased pressure and temperature), and a jet nozzle uses that energy to create a high velocity at the engine exhaust.

[1] The Aircraft Gas Turbine Engine and Its Operation, Section 1, United Technologies Corporation, Part No. P&W 182408 (1988) (Exhibit 12 to my Declaration), page 1-14.

3. According to Newton's Second Law, the thrust generated by the engine can be expressed as $T = \dot{M}(V_2 - V_1)$. $\dot{M}$ is the mass flow rate of air through the engine, V_1 is the velocity of the air entering the engine air inlet, and V_2 is the velocity of the air exiting the jet nozzle. Strictly speaking, the mass flow rate $\dot{M}$ also includes the fuel, but the mass flow rate of the fuel is much smaller than the mass flow rate of the air.

4. Since thrust $T = \dot{M}(V_2 - V_1)$, more thrust can be generated if the mass flow rate is increased by taking more air into the engine. This is the principle behind a turbofan (or "ducted fan gas turbine") engine. Figure 2 illustrates a typical such engine.

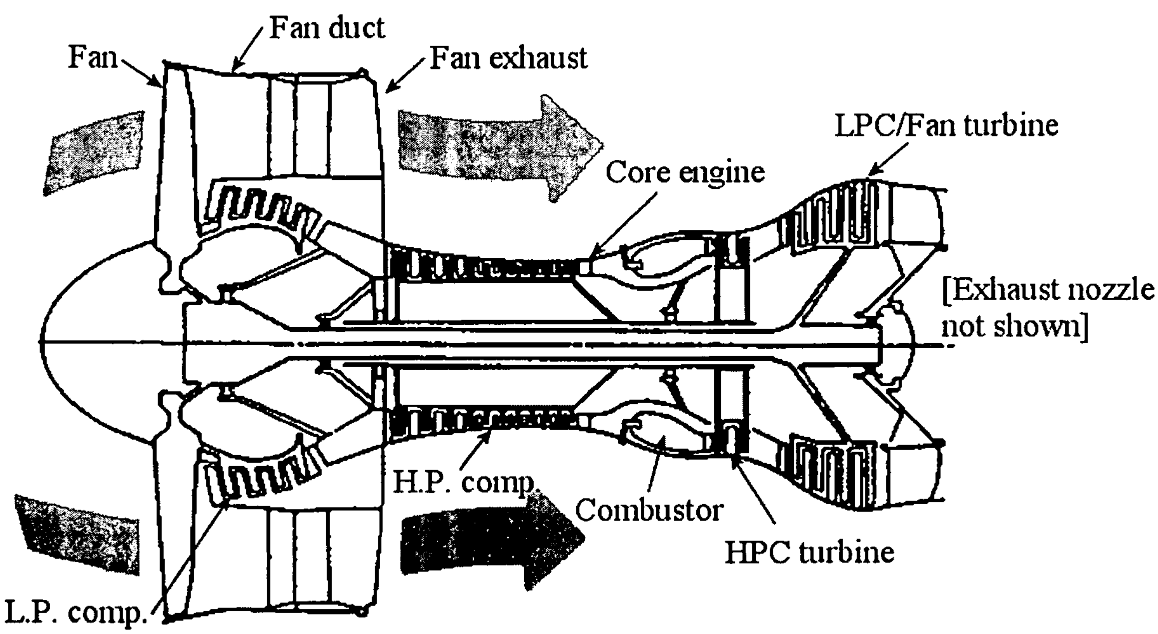


FIGURE 2[2] - Typical Turbofan Engine

5. The fan is essentially a large compressor in a duct ahead of the intake of a core turbojet engine like that shown in Figure 1. The structure of the core engine in Figure 2 is a little different from that shown in Figure 1. The core engine in Figure 2 has a low pressure compressor (L.P.C.) at the engine air inlet followed by a separate high pressure compressor

[2] Adapted from The Aircraft Gas Turbine Engine and Its Operation, supra, page 1-17.

(H.P.C.). In Figure 2's core engine the combustor (Figure 1's combustion chamber) is followed by an HPC turbine that extracts some of the energy from the combustion products to drive the high pressure compressor through a first shaft. An LPC/Fan turbine drives the fan and the low pressure compressor through a second shaft coaxial with the first shaft. The combustion products leaving the second turbine pass through a jet exhaust nozzle (omitted from this depiction) as discussed above. In other words, Figure 2's core engine has some different components, but the engine operating principles are the same as the basic turbojet engine in Figure 1.

6. The fan is what differentiates the engine in Figure 2 from the turbojet engine in Figure 1. The fan takes in additional air, thus increasing $\dot{M}$ in the thrust equation above. Some of the air taken in through the fan enters the core engine (at the low pressure compressor inlet), and is compressed, burned, and exhausted as in a basic turbojet engine. But a large quantity of the air bypasses the core engine and exits through the fan exhaust, which is configured to optimize the exit air velocity. The fan duct may either end shortly downstream of the fan as shown in Figure 2, or extend the length of the core engine so that the flow from the fan exits with the core engine exhaust. Turbofan engines are generally more efficient and quieter than turbojet engines, and consequently are used on the overwhelming majority of commercial aircraft today.

7. The fan in a turbofan engine can be thought of as a cross between a propeller and an axial flow compressor. It functions like a propeller in that it moves large quantities of air, but it has a larger number of blades than a propeller and they are arrayed around a hub like an axial flow compressor of a turbojet engine. Figure 3 shows how blades are arranged in the fan of a typical turbofan engine:

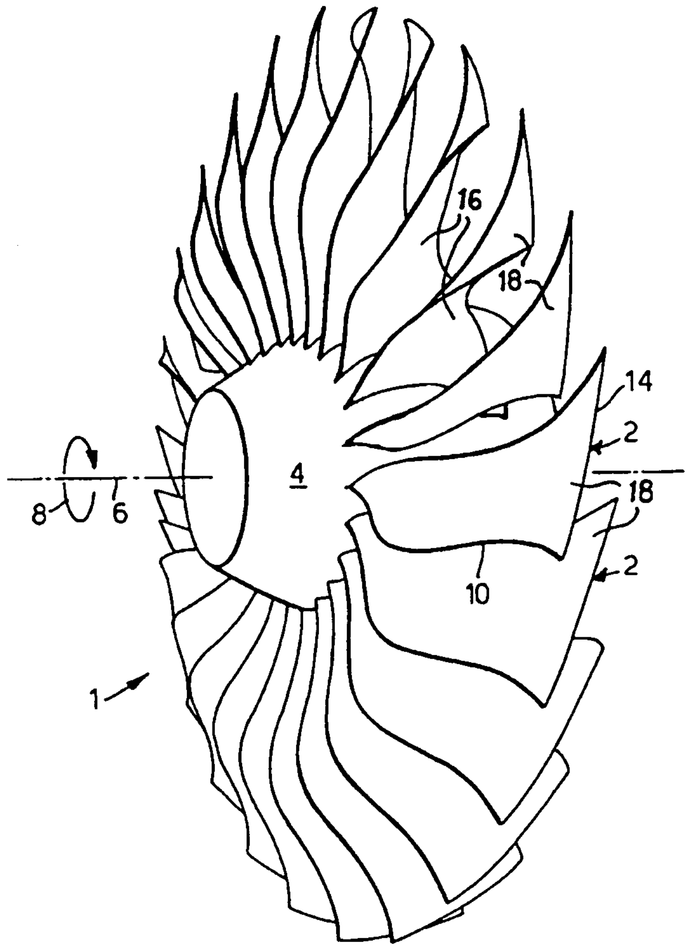


FIGURE 3[3] - Fan Blades Around a Hub

8. The blades 2 are attached to a hub 4, which rotates in the direction of arrow 8. This figure makes it clear that in a rotating fan the tangential velocity (that is, the velocity in the direction of rotation, or tangential to the arrow 8) of any given point on a blade increases radially outwardly along the blade from the root (at the hub 4) to the tip 14. In most turbofan engines today, there is a location along the blade at which the blade velocity, measured relative to the air approaching the fan inlet, reaches the speed of sound. In turn, that presents special problems, addressed by the UTC and Rolls patents.

[3] Figure 1 of the Rolls '077 Patent (Exhibit 5 to my Declaration). The turbofan engine in the Rolls '077 Patent has a fan duct (see patent Figure 5a, for example), but the duct is omitted from patent Figure 1.

9. To understand those problems it is necessary first to understand the physics of fluid flow over a body traveling in excess of the speed of sound. The familiar term "supersonic" is used to describe flow of a body (say, a gas turbine engine fan blade) at a velocity faster than the speed of sound in the fluid through which it is moving (in this case air).

10. Any object moving through a fluid creates pressure waves. Sound, being a pressure wave, thus can travel only as fast as the speed at which pressure waves can propagate through the fluid. Figure 4 below represents the flow field resulting from a point moving through air. The point parts the air as it moves, and as it does, the disturbances resulting from the point parting the air manifest themselves as pressure waves. Each pressure wave takes the form of an expanding spherical wave front starting at the point itself at each location along the path of movement. This wave propagates outwardly from the point exactly at the speed of sound V_s in the fluid through which the point is traveling (in this case air).

11. Figure 4 shows the flow field when the point is at location S, with previous locations at finite time intervals depicted at S', S", etc. While there are actually an infinite number of such locations, this simplified depiction enables consideration of the effect of the point's velocity U by illustrating the pressure waves created at a few discrete locations of the moving point. That is, Figure 4 is a snapshot of a theoretical flow field at the time when the point has reached location S, showing the current position of the pressure waves (disturbances) that were created when the point was at S', S", etc.

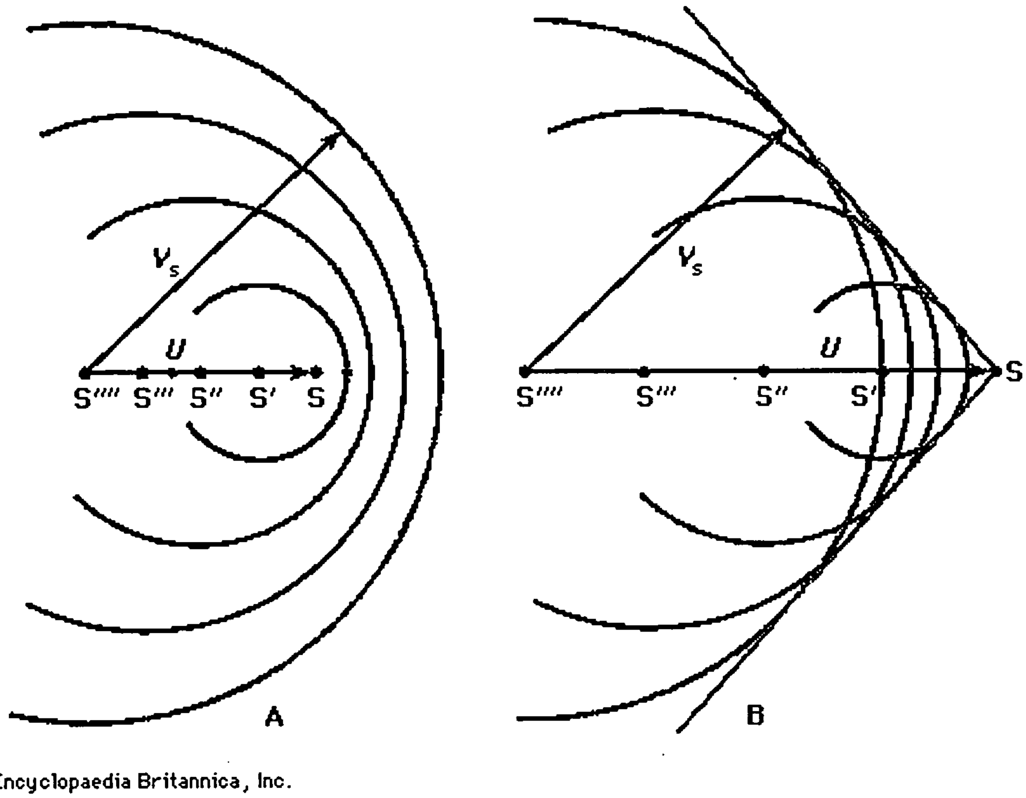


FIGURE 4[4] - Theoretical Flow Fields Caused by a Moving Point
(A: Point traveling at $U < V_s$)
(B: Point traveling at $U > V_s$)

12. The depiction in Figure 4 is attributed to Austrian physicist Ernst Mach (1838-1916)[5], who gave his name to "Mach number" (M), which is the ratio of the velocity U of an object traveling through a medium to the speed of sound V_s in the medium ($M = U/V_s$). "A" is the case where the velocity U of the point is less than the speed of sound, or subsonic ($M < 1$). Because the pressure waves propagate at V_s, which is greater than U in this case, the pressure waves can stay ahead of the moving point as shown in "A." However, "B" shows the case when the point's velocity U is greater than the speed of sound V_s ($M > 1$). Thus the pressure waves, which by definition can propagate at a velocity no greater than the speed of sound V_s (as discussed above), no longer can stay ahead of the moving point.

---

[4] "Compressible Flow in Gases" Encyclopædia Britannica, 2002 (Exhibit 13 to my Declaration), Figure 8, page 4.
[5] "Mach, Ernst" Encyclopædia Britannica, 2002 (Exhibit 14 to my Declaration).

13. Figure 4 is a theoretical depiction that has been found useful in analyzing various aspects of supersonic flow. One thing that makes it theoretical is that S is an infinitesimally small geometric point, rather than an actual physical object. Accordingly, Figure 4's "pressure waves" are actually infinitesimally small pressure disturbances (perturbations). In addition, since the point is moving continuously, the "spaces" between successive locations S are also infinitesimal, resulting in an infinite number of the depicted small perturbation pressure waves. That is, Figure 4 only shows individual spherical perturbations generated at a few discrete intervals during the point's travel. There would actually be an infinite number of the perturbations, which together would form a Mach cone that can be drawn tangent to the circles (actually spheres) in Figure 4. This theoretical Mach cone, illustrated in Figure 5, has a characteristic Mach angle $\alpha = \sin^{-1}(V_s/U)$ ($= \sin^{-1}(1/M)$) relative to the path of the point S.

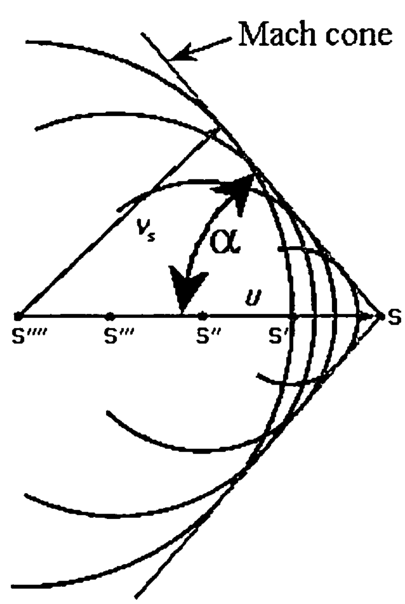


FIGURE 5 - Mach Cone Orientation

14. If the point in Figures 4 and 5 is an actual physical object, the Mach cone becomes a shock front, also referred to as a shock wave, because the perturbations then become actual pressure waves. A shock wave is actually a nearly instantaneous pressure change, as air ahead of

the shock wave traveling at a supersonic velocity relative to the point rapidly decelerates as it passes the shock wave. Shock waves are the cause of so-called "sonic booms" caused by aircraft traveling at supersonic velocities under certain conditions. The almost instantaneous velocity change creates a correspondingly rapid change in pressure, which in the case of a sonic boom is experienced by an observer on the ground as a loud noise (which is nothing more than how the human ear perceives a rapid pressure change).[6]

15. Of course, in any practical case, such as flow over an aircraft or through the fan of a turbofan gas turbine engine, the actual situation is more complex than suggested by Figures 4 and 5. But regardless of the complexity of the flow field, the passage of a flow through shock waves involves energy losses, and engineering efforts where supersonic flow is involved often attempt to reduce energy losses caused by the presence of attendant shock waves. It is also true that shock waves can be weaker or stronger depending on the actual Mach number of the approaching flow. The higher the approaching flow velocity, the more energy loss is involved when the flow suddenly decelerates by passing through a shock wave.

16. That is one of the reasons that supersonic aircraft have highly swept wings, meaning that the wing leading edge presents an angle to the approaching airflow. This is illustrated in the following Figure 6, adapted from Figure 1A of U.S. Patent 3,989,406 to Bliss (Exhibit 8 to my Declaration).

---

[6] Contrary to the belief of some, a sonic boom does not occur because an aircraft "breaks the sound barrier" when it accelerates through the speed of sound. A sonic boom is the sound heard as a shock wave from an aircraft traveling at $M > 1$ sweeps the ground, and it is heard by all observers on the ground under the aircraft's flight path. That is, two observers at different locations will hear a sonic boom at two different times. The idea of a physical "sound barrier" resulted from an incomplete understanding of supersonic flow at the time of early attempts to fly faster than the speed of sound. It was later confirmed that when an aircraft designed for subsonic flight approaches $M = 1$, flow over the aircraft can become highly unstable and cause vibrations of magnitudes the airframe cannot withstand. Resulting catastrophic aircraft failures led some to believe that supersonic flight was physically impossible.

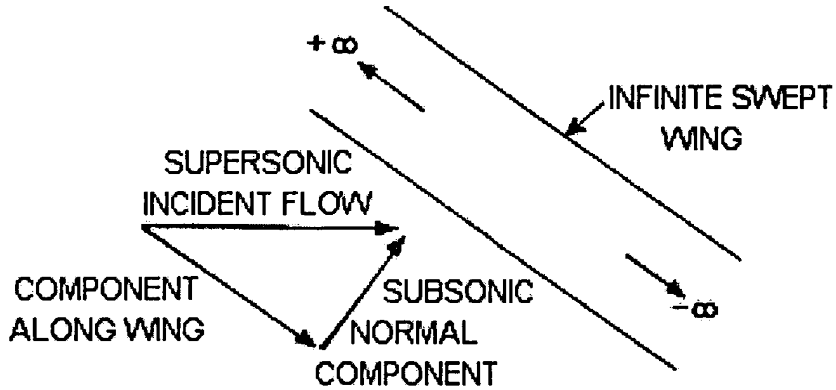


FIGURE 6 - Flow Over a Swept Wing (Plan View)

17. This schematic depiction of an aircraft wing shows how sweep reduces the velocity of the supersonic incident flow when it is resolved into a component normal to the wing leading edge and a component along the wing span. (The wing is depicted unattached to an aircraft and as being infinitely long so that flow effects caused by the aircraft fuselage and wing tips can be ignored.) The wing in Figure 6 has enough sweep to reduce the flow velocity normal to the leading edge to $M < 1$, thus avoiding shock losses altogether. However, the principle holds even when the normal flow is supersonic: the strength of the shock wave, and thus the amount of the energy loss, is reduced when the wing is swept.

18. The same principle is involved in the present invention, namely sweeping the fan blade leading edge to weaken the shock waves and thereby reduce energy losses in the flow through the fan. It is not surprising that sweeping the blade leading edge occurred to fan engineers, since a fan blade is an airfoil that performs in many ways analogous to an airplane wing. However, the shock wave system in supersonic flow through a rotating fan is more complex than it is in supersonic flow over an aircraft wing.

19. Among those complications is that the shock waves from one blade influence the shape and location of the shock waves from neighboring blades. In addition, the presence of the

duct creates complex flow conditions that affect the shape and location of the shock waves. Minor changes in the cross-sectional shape of the airfoil comprising the blades also strongly influences the orientation and strength of the shock system created when the fan is rotating fast enough that the blades' leading edges are traveling at supersonic velocities relative to the approaching air. A fan engineer must also decide on the number of blades to incorporate in the fan, and that also affects the nature of the shock system in the flow.

20. There are also numerous physical constraints that any fan engineer must meet when designing a fan for a turbofan engine. One of them relates to the three-dimensional shape of the blade and is illustrated in Figure 7, which is adapted from Figure 2 of U.S. Patent 4,012,172 to Schwaar et al. (Exhibit 6 to my Declaration).

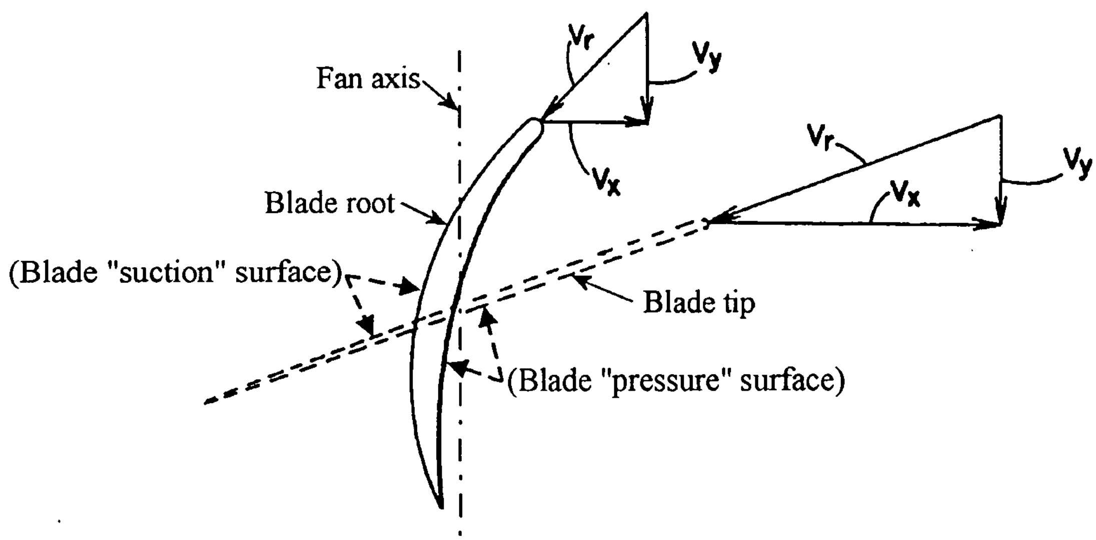


FIGURE 7 - Fan Blade Twist from Root to Tip

21. This is a view of a fan blade looking radially inward; the blade at the root is shown in solid lines and the blade at the tip is in dashed lines. The incoming air velocity V_y is along the fan axis, and under normal operating conditions both its direction and magnitude are essentially constant from the blade root to the tip. As noted in paragraph 18, a fan blade functions in many respects like an airplane wing, in that the blade is an airfoil and flow over the blade thus creates a

higher pressure on one side, sometimes called the blade's "pressure" surface, and a lower pressure on the other, sometimes called the blade's "suction" surface. (In an airplane this pressure differential between opposite sides of the wing causes lift.) The blade tangential velocity (in the direction of rotation of the fan) is V_x, and as discussed in paragraph 8, this velocity component necessarily is greater at the blade tip than at the root. Accordingly, the velocity V_r of the flow approaching the blade changes in magnitude <u>and</u> direction along the blade span. If the blade is to function at all, it must twist about a longitudinal axis (coming out of the paper) in order to be at the proper angle to the approaching airflow at all points along the blade span. Otherwise, the flow will separate from the blade suction surface, preventing the blade from generating the necessary pressure differential between the suction and pressure surfaces.

22. Fan blades must also be as light as possible to maximize aircraft payload, but they still have to withstand stresses created by fan rotational speeds of two or three thousand revolutions per minute. Accordingly, a fan engineer must configure the blade to minimize those stresses. Furthermore, fan blades must withstand impact by foreign objects such as birds ingested by the engine during flight. Accordingly, the fan engineer must find the best possible blade configuration that satisfies all of those geometric constraints while still meeting the performance requirements demanded by the aircraft with which the engine is to be used.

23. All of these constraints frustrated most prior attempts to provide practicable swept fan blades. One approach in the 1970s and into the 1980s introduced sufficient sweep at the blade leading edge to avoid shocks altogether. An example of that approach is shown in the following Figure 8, which reproduces Bliss's Figure 2.

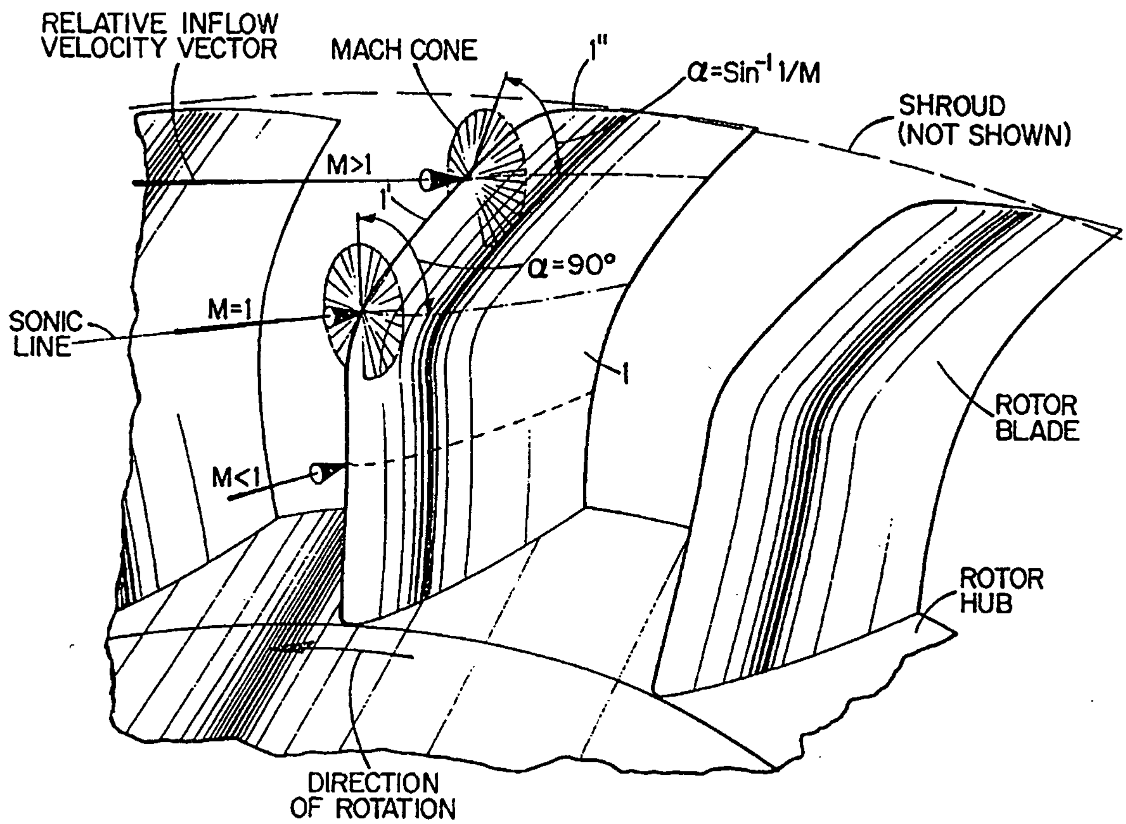


FIGURE 8 - Prior Art Approach to Reduce Shock Losses

24. This blade leading edge is swept to a degree that places it behind the "Mach cone" (see Figure 5). The object of introducing such a large degree of sweep was to avoid shock waves in the first place, which would allow the fan engineer to avoid the complications they cause.

25. Using severe sweep in an attempt to eliminate shock waves proved impracticable for a number of reasons. A blade having a configuration theoretically usable from an aerodynamic standpoint and strong enough to withstand stresses usually weighed so much that it offset any potential gains in fan efficiency gained by sweeping the leading edge. That effect was exacerbated for high aspect ratio (long and narrow) blades like those used in large commercial aircraft, which is and was a primary application for supersonic fan blades. In addition, blades of this general configuration still did not perform aerodynamically as well as expected, for a variety

of complex causes. Blade leading edges swept to a lesser degree, or with configurations incorporating sweep in some other manner, were also considered. But blades with leading edge sweep did not gain widespread acceptance from fan engineers because all such blades of which I am aware proved aerodynamically or mechanically less desirable than more conventional blades for one reason or another.

26. By 1995, one swept blade configuration being widely studied had a leading edge with a forward swept inner region and a rearward swept region outward of the inner region, like the blades shown in Figure 2 of the UTC '985 Patent (Exhibit 3 to my Declaration) and Figure 5a of the Rolls '077 Patent (Exhibit 5 to my Declaration). The rearward sweep mitigates shock losses, as discussed above; forward sweep in the inner region makes the rearward sweep practicable from the standpoint of mechanical stresses on the blade. Such a blade is also shown in earlier patents, such as Schwaar and U.S. Patent 4,726,737 to Weingold et al. (Exhibit 7 to my Declaration). These blades, especially in high-performance fans, can create shock waves upstream of the blade at the duct wall. Fan engineers knew that such shocks decrease aerodynamic efficiency and the range of operating conditions under which the blade is aerodynamically stable, but were unable to design a fan blade that used a swept leading edge to improve efficiency and was still stable under all engine operating conditions.

US005642985A

3

United States Patent [19]

Spear et al.

[11] **Patent Number: 5,642,985**

[45] **Date of Patent: Jul. 1, 1997**

[54] **SWEPT TURBOMACHINERY BLADE**

[75] Inventors: **David A. Spear**, Manchester; **Bruce P. Biederman**, Meriden, both of Conn.; **John A. Orosa**, Palm Beach Gardens, Fla.

[73] Assignee: **United Technologies Corporation**, Hartford, Conn.

[21] Appl. No.: **559,965**

[22] Filed: **Nov. 17, 1995**

[51] Int. Cl.^{6} **F01D 5/14**

[52] U.S. Cl. **416/238**; 415/181; 416/242

[58] **Field of Search** 415/181, 220; 416/238, 242, 243

[56] **References Cited**

U.S. PATENT DOCUMENTS

1,964,525	6/1934	McMahan	170/159
2,154,313	4/1939	McMahan	230/274
2,915,238	12/1959	Szydlowski	230/134
2,934,259	4/1960	Hausmann	415/181
2,935,246	5/1960	Roy	415/181
3,416,725	12/1968	Bohanon	230/259
3,444,817	5/1969	Caldwell	103/88
3,692,425	9/1972	Erwin	415/181
3,989,406	11/1976	Bliss	415/1
4,012,172	3/1977	Schwaar et al.	416/228
4,358,246	11/1982	Hanson et al.	416/223 R
4,408,957	10/1983	Kurzrock et al.	415/181
4,726,737	2/1988	Weingold et al.	416/223 A
4,737,077	4/1988	Vera	416/242
4,784,575	11/1988	Nelson et al.	416/226
5,112,192	5/1992	Weetman	416/201 A
5,167,489	12/1992	Wadia et al.	415/182.1

FOREIGN PATENT DOCUMENTS

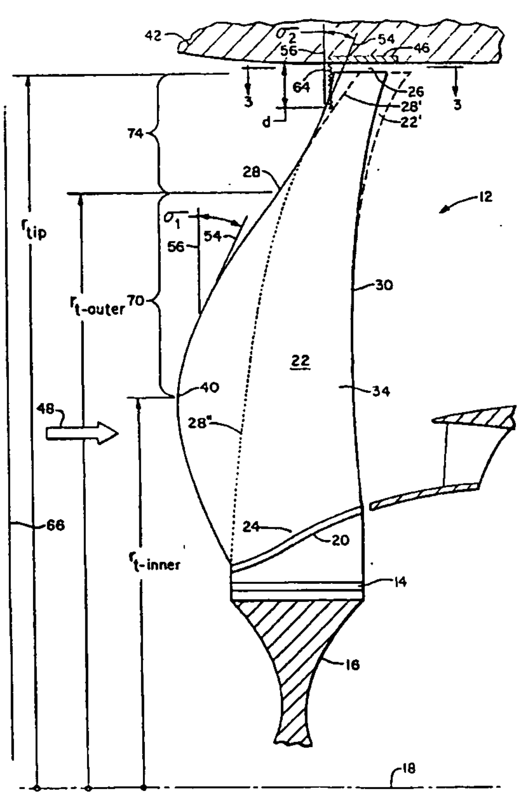
1528965	12/1989	U.S.S.R.	416/242

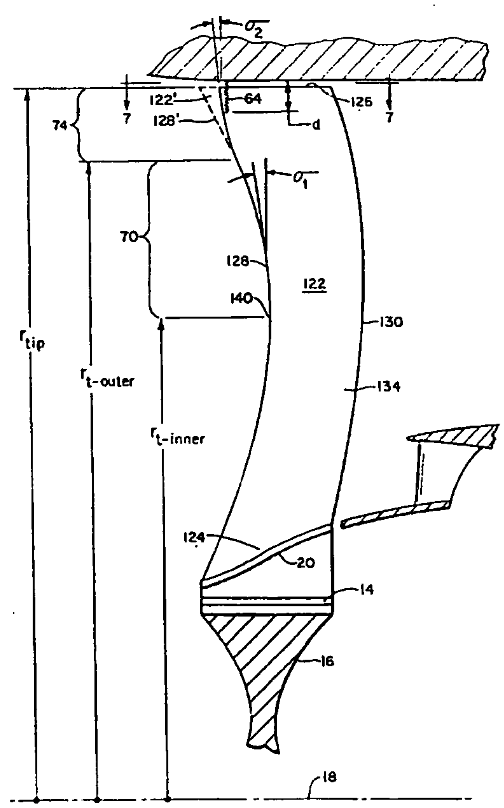
Primary Examiner—Edward K. Look
Assistant Examiner—Mark Sgantzos
Attorney, Agent, or Firm—Kenneth C. Baran

[57] **ABSTRACT**

A swept turbomachinery blade for use in a cascade of such blades is disclosed. The blade (12) has an airfoil (22) uniquely swept so that an endwall shock (64) of limited radial extent and a passage shock (66) are coincident and a working medium (48) flowing through interblade passages (50) is subjected to a single coincident shock rather than the individual shocks. In one embodiment of the invention the forwardmost extremity of the airfoil defines an inner transition point (40) located at an inner transition radius r_t-inner. The sweep angle of the airfoil is nondecreasing with increasing radius from the inner transition radius to an outer transition radius $r_{t\text{-}outer}$, radially inward of the airfoil tip (26), and is nonincreasing with increasing radius between the outer transition radius and the airfoil tip.

3 Claims, 7 Drawing Sheets





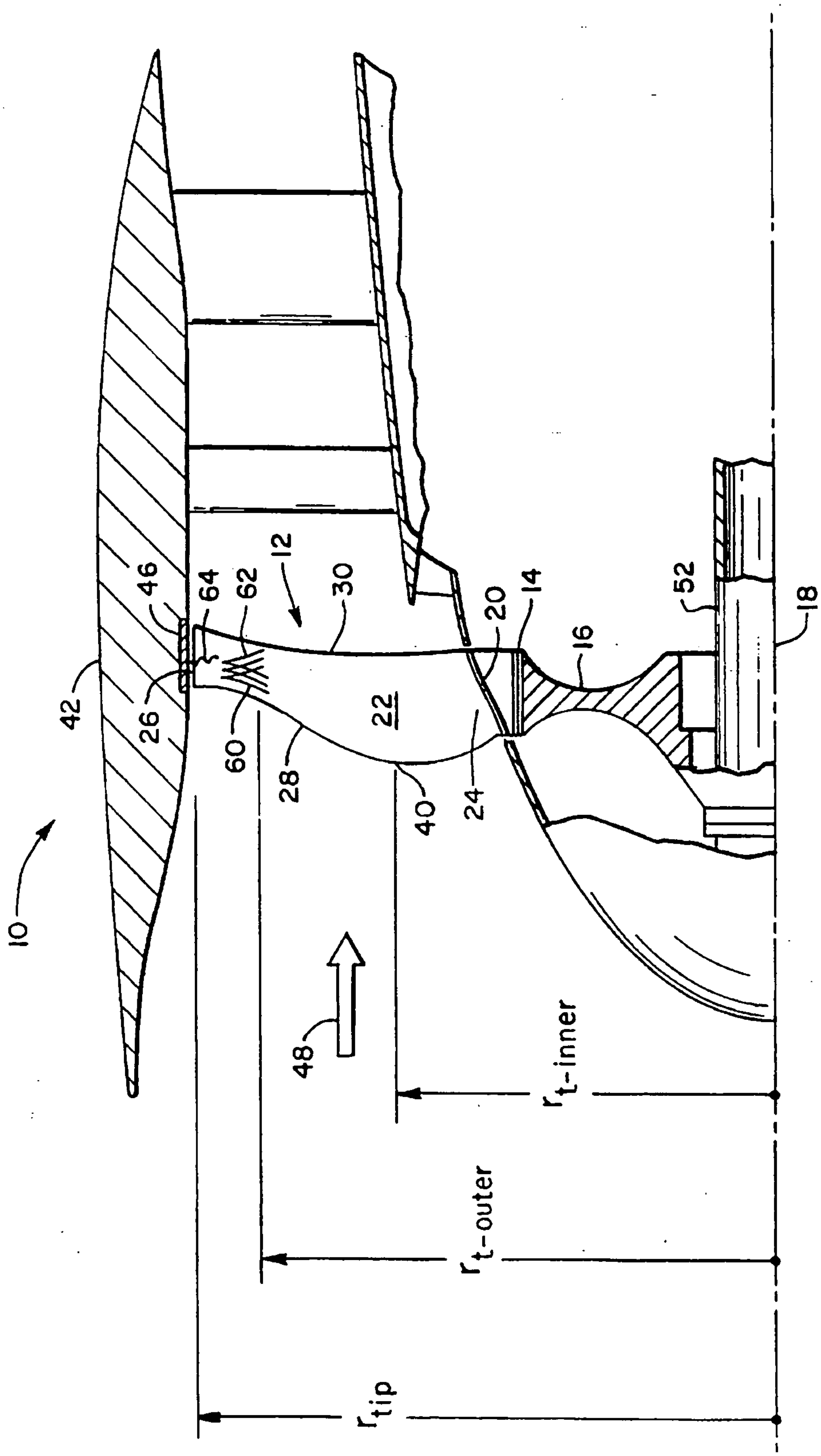
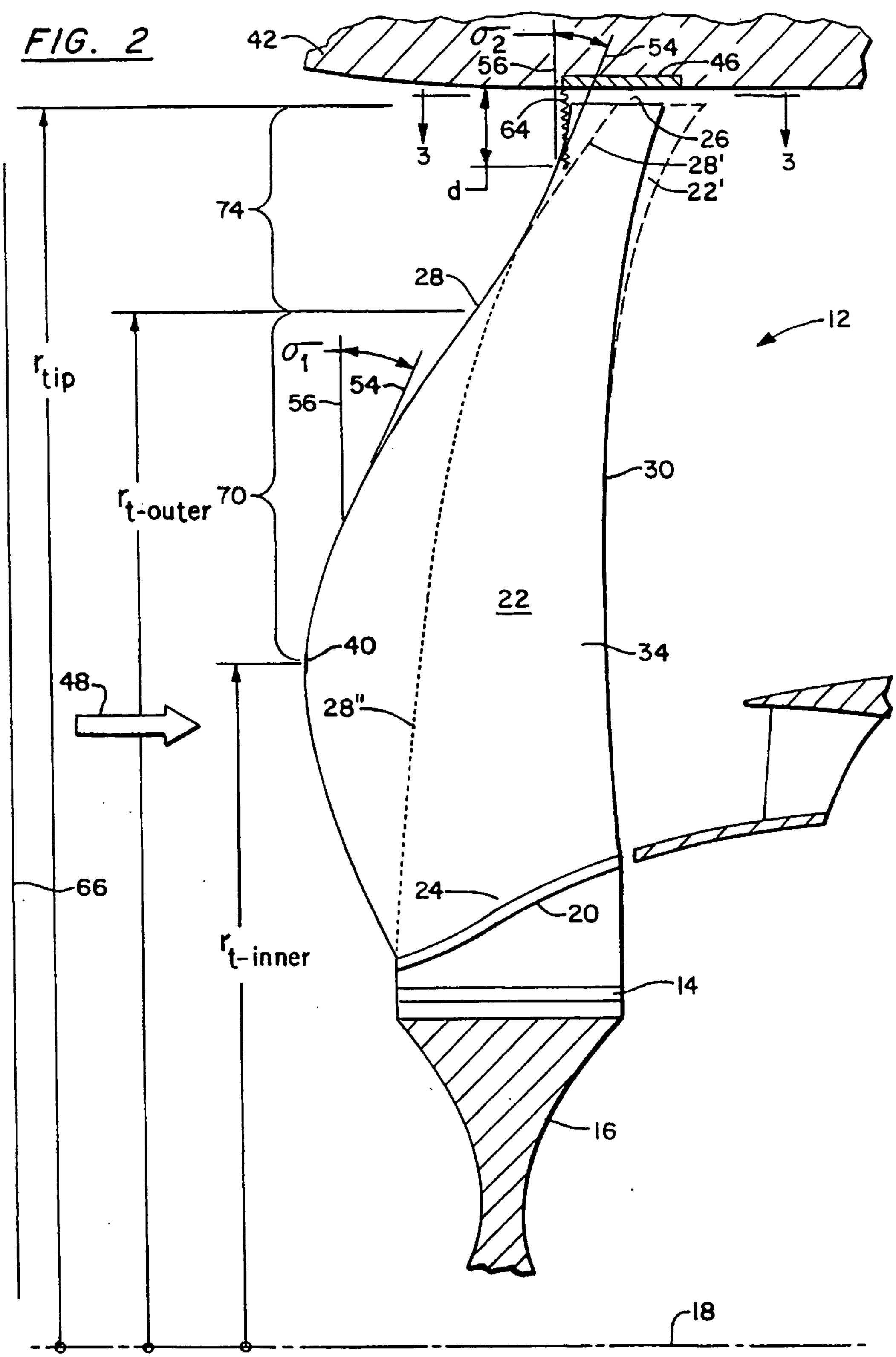

10
12
14
16
18
20
22
24
26
28
30
40
42
46
48
52
60
62
64
$r_{t\text{-inner}}$
$r_{t\text{-outer}}$
r_{tip}

FIG. 1

FIG. 2



42
σ_2
54
56
46
3
64
26
3
28'
d
22'
74
28
12
σ_1
54
56
r_{tip}
30
$r_{t\text{-}outer}$
70
22
40
34
48
28"
66
24
20
$r_{t\text{-}inner}$
14
16
18

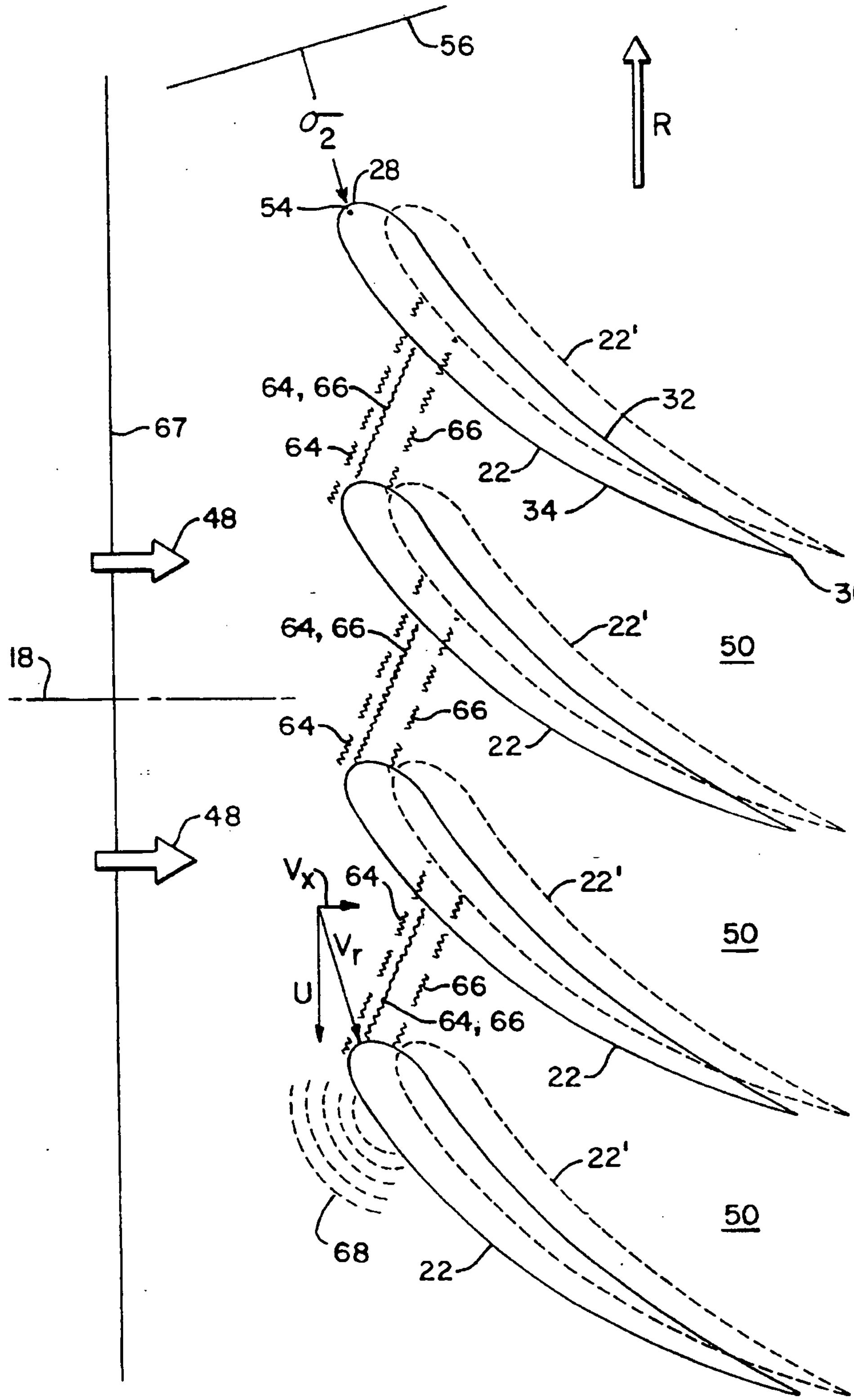

56
σ_2
R
28
54
22'
64, 66
32
67
64
66
22
34
48
30
22'
64, 66
50
18
64
66
22
48
22'
V_x
64
50
V_r
66
U
64, 66
22
22'
50
68
22

FIG. 3

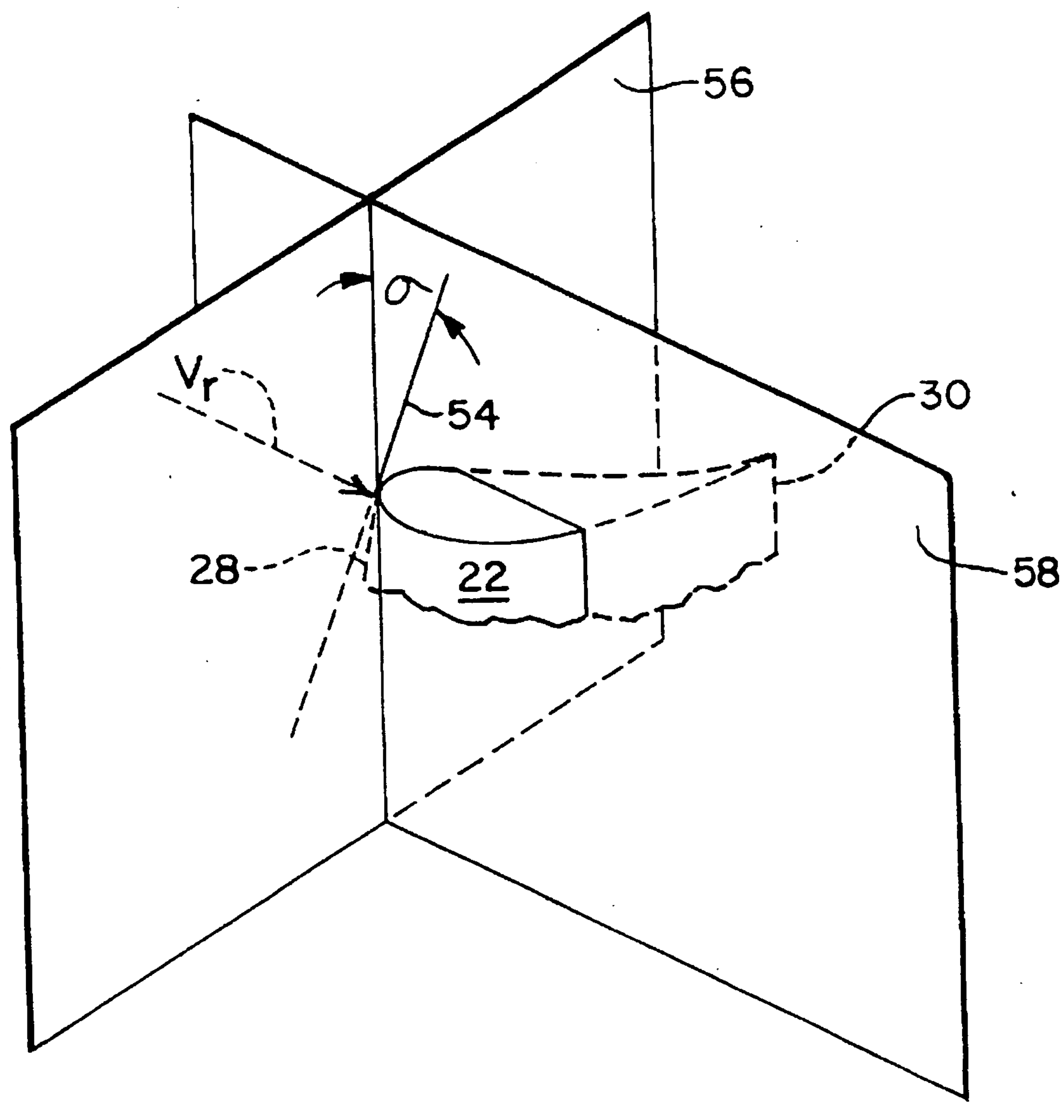


FIG. 4

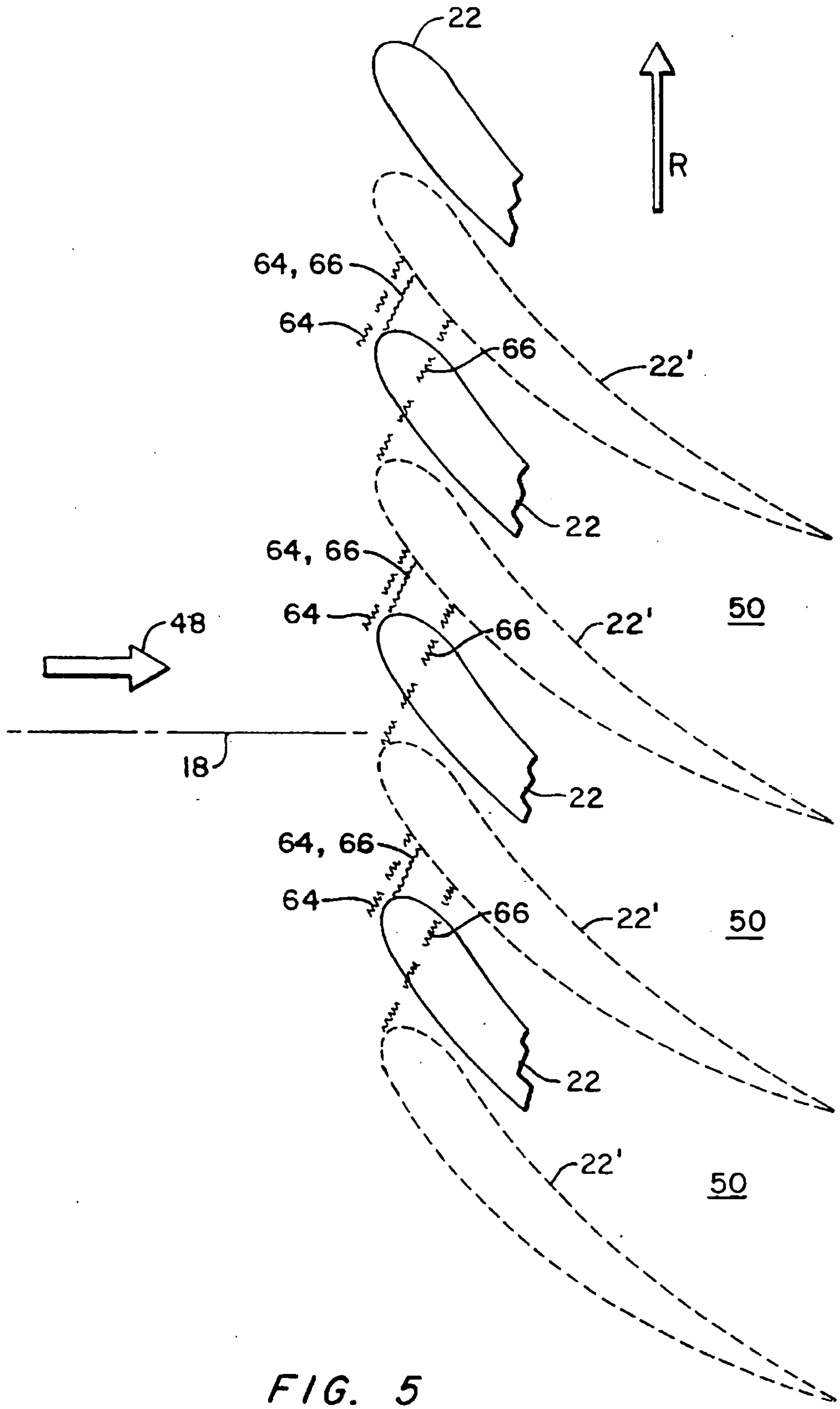
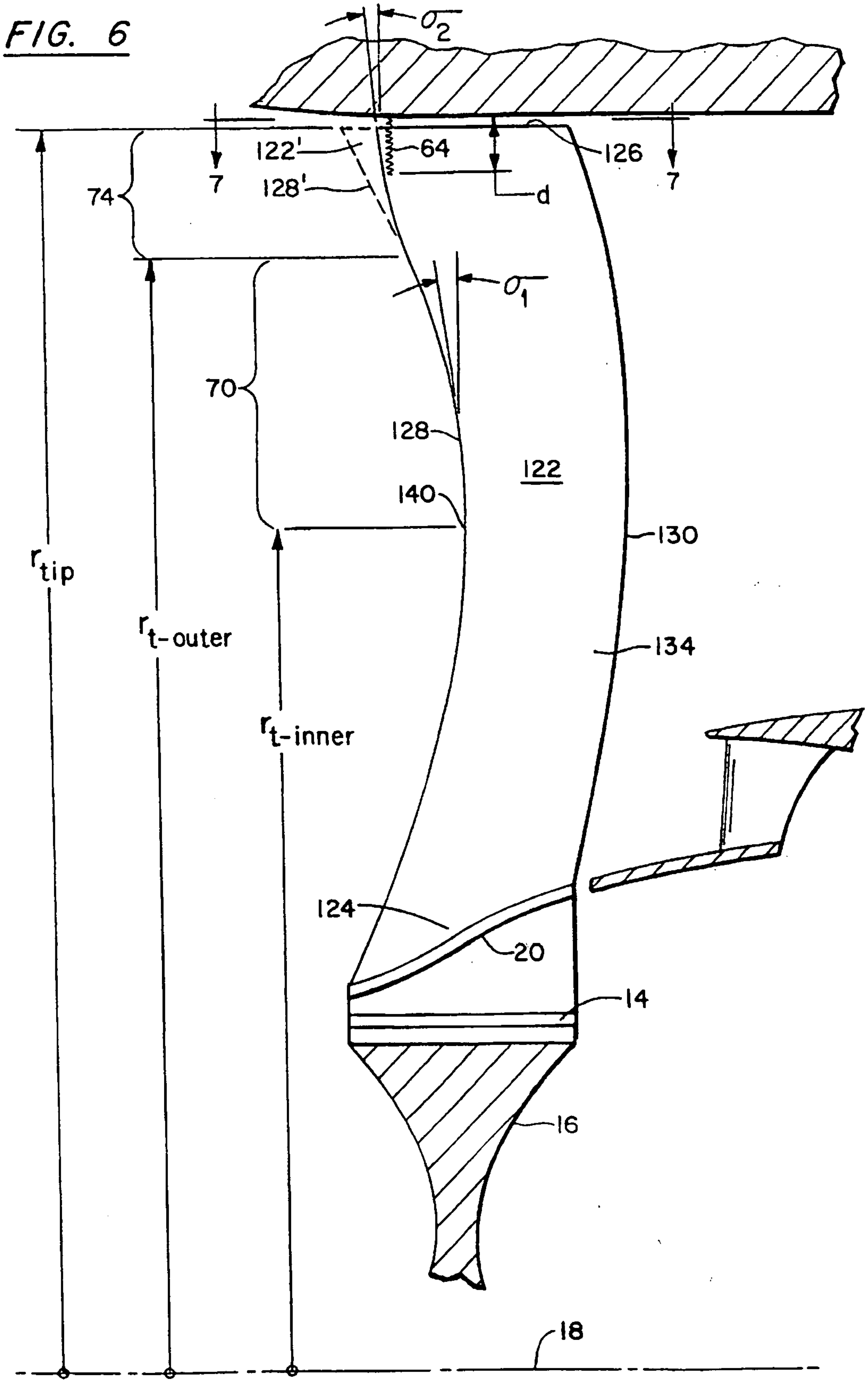

22
R
64, 66
64
66
22'
22
64, 66
64
48
66
22'
50
18
22
64, 66
64
66
22'
50
22
22'
50

FIG. 5

FIG. 6



σ_2
122'
64
126
7
7
74
128'
d
σ_1
70
128
122
140
130
r_{tip}
$r_{t-outer}$
134
$r_{t-inner}$
124
20
14
16
18

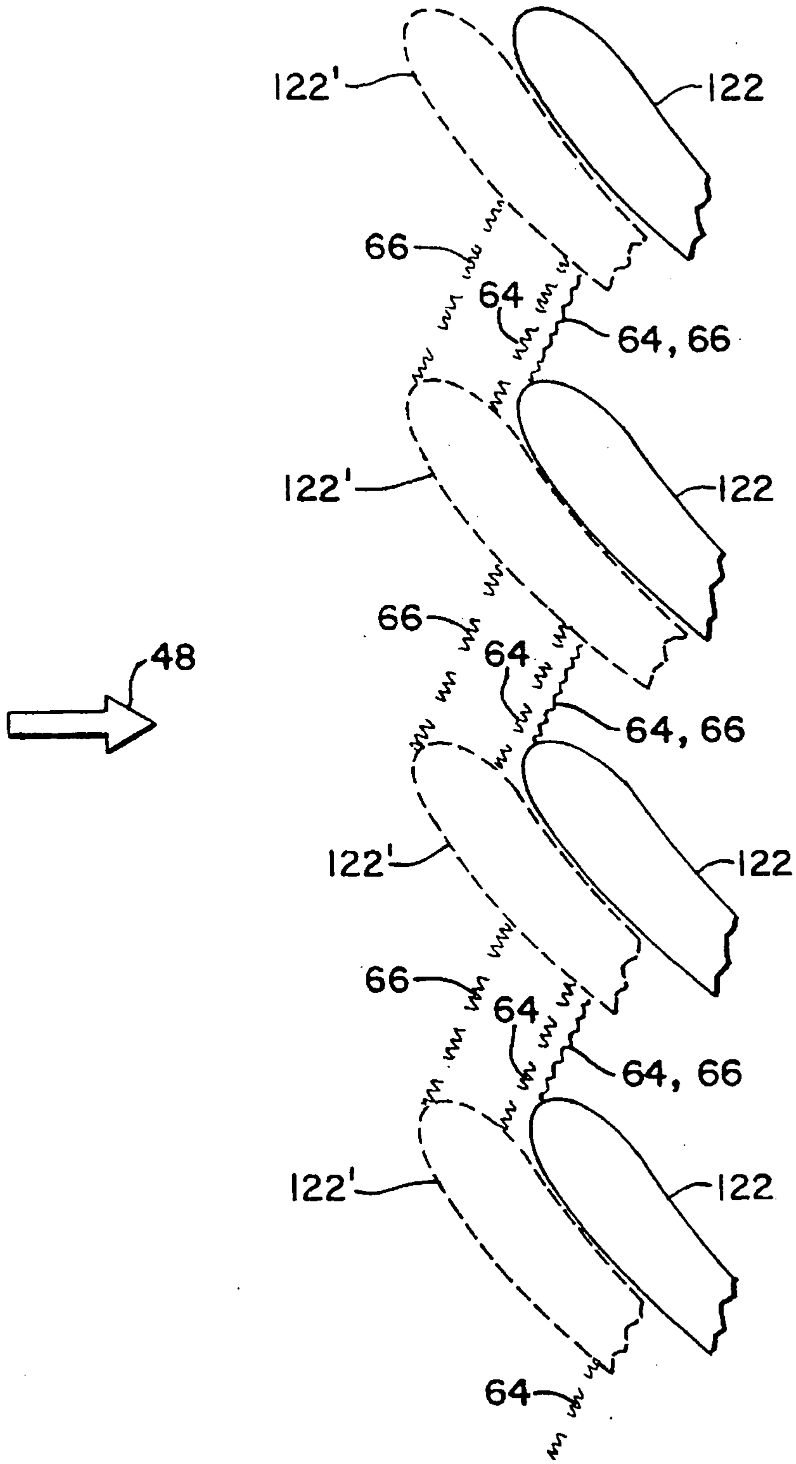

122'
122
66
64
64, 66
122'
122
66
64
48
64, 66
122'
122
66
64
64, 66
122'
122
64

FIG. 7

SWEPT TURBOMACHINERY BLADE

TECHNICAL FIELD

This invention relates to turbomachinery blades, and particularly to blades whose airfoils are swept to minimize the adverse effects of supersonic flow of a working medium over the airfoil surfaces.

BACKGROUND OF THE INVENTION

Gas turbine engines employ cascades of blades to exchange energy with a compressible working medium gas that flows axially through the engine. Each blade in the cascade has an attachment which engages a slot in a rotatable hub so that the blades extend radially outward from the hub. Each blade has a radially extending airfoil, and each airfoil cooperates with the airfoils of the neighboring blades to define a series of interblade flow passages through the cascade. The radially outer boundary of the flow passages is formed by a case which circumscribes the airfoil tips. The radially inner boundary of the passages is formed by abutting platforms which extend circumferentially from each blade.

During engine operation the hub, and therefore the blades attached thereto, rotate about a longitudinally extending rotational axis. The velocity of the working medium relative to the blades increases with increasing radius. Accordingly, it is not uncommon for the airfoil leading edges to be swept forward or swept back to mitigate the adverse aerodynamic effects associated with the compressibility of the working medium at high velocities.

One disadvantage of a swept blade results from pressure waves which extend along the span of each airfoil suction surface and reflect off the surrounding case. Because the airfoil is swept, both the incident waves and the reflected waves are oblique to the case. The reflected waves interact with the incident waves and coalesce into a planar aerodynamic shock which extends across the interblade flow channel between neighboring airfoils. These "endwall shocks" extend radially inward a limited distance from the case. In addition, the compressibility of the working medium causes a passage shock, which is unrelated to the above described endwall shock, to extend across the passage from the leading edge of each blade to the suction surface of the adjacent blade. As a result, the working medium gas flowing into the channels encounters multiple shocks and experiences unrecoverable losses in velocity and total pressure, both of which degrade the engine's efficiency. What is needed is a turbomachinery blade whose airfoil is swept to mitigate the effects of working medium compressibility while also avoiding the adverse influences of multiple shocks.

DISCLOSURE OF THE INVENTION

It is therefore an object of the invention to minimize the aerodynamic losses and efficiency degradation associated with endwall shocks by limiting the number of shocks in each interblade passage.

According to the invention, a blade for a blade cascade has an airfoil which is swept over at least a portion of its span, and the section of the airfoil radially coextensive with the endwall shock intercepts the endwall shock extending from the neighboring airfoil so that the endwall shock and the passage shock are coincident.

In one embodiment the axially forwardmost extremity of the airfoil's leading edge defines an inner transition point located at an inner transition radius radially inward of the airfoil tip. An outer transition point is located at an outer transition radius radially intermediate the inner transition radius and the airfoil tip. The outer transition radius and the tip bound a blade tip region while the inner and outer transition radii bound an intermediate region. The leading edge is swept at a first sweep angle in the intermediate region and is swept at a second sweep angle over at least a portion of the tip region. The first sweep angle is generally nondecreasing with increasing radius and the second sweep angle is generally non-increasing with increasing radius.

The invention has the advantage of limiting the number of shocks in each interblade passage so that engine efficiency is maximized.

BRIEF DESCRIPTION OF THE DRAWINGS

FIG. 1 is a cross sectional side elevation of the fan section of a gas turbine engine showing a swept back fan blade according to the present invention.

FIG. 2 is an enlarged view of the blade of FIG. 1 including an alternative leading edge profile shown by dotted lines and a prior art blade shown in phantom.

FIG. 3 is a developed view taken along the line 3—3 of FIG. 2 illustrating the tips of four blades of the present invention along with four prior art blades shown in phantom.

FIG. 4 is a schematic perspective view of an airfoil fragment illustrating the definition of sweep angle.

FIG. 5 is a developed view similar to FIG. 3 illustrating an alternative embodiment of the invention and showing prior art blades in phantom.

FIG. 6 is a cross sectional side elevation of the fan section of a gas turbine engine showing a forward swept fan blade according to the present invention and showing a prior art fan blade in phantom.

FIG. 7 is a developed view taken along the line 7—7 of FIG. 6 illustrating the tips of four blades of the present invention along with four prior art blades shown in phantom.

BEST MODE FOR CARRYING OUT THE INVENTION

Referring to FIGS. 1–3, the forward end of a gas turbine engine includes a fan section **10** having a cascade of fan blades **12**. Each blade has an attachment **14** for attaching the blade to a disk or hub **16** which is rotatable about a longitudinally extending rotational axis **18**. Each blade also has a circumferentially extending platform **20** radially outward of the attachment. When installed in an engine, the platforms of neighboring blades in the cascade abut each other to form the cascade's inner flowpath boundary. An airfoil **22** extending radially outward from each platform has a root **24**, a tip **26**, a leading edge **28**, a trailing edge **30**, a pressure surface **32** and a suction surface **34**. The axially forwardmost extremity of the leading edge defines an inner transition point **40** at an inner transition radius r_t-inner, radially inward of the tip. The blade cascade is circumscribed by a case **42** which forms the cascade's outer flowpath boundary. The case includes a rubstrip **46** which partially abrades away in the event that a rotating blade contacts the case during engine operation. A working medium fluid such as air **48** is pressurized as it flows axially through interblade passages **50** between neighboring airfoils.

The hub **16** is attached to a shaft **52**. During engine operation, a turbine (not shown) rotates the shaft, and therefore the hub and the blades, about the axis **18** in direction R. Each blade, therefore, has a leading neighbor

which precedes it and a trailing neighbor which follows it during rotation of the blades about the rotational axis.

The axial velocity V_x (FIG. 3) of the working medium is substantially constant across the radius of the flowpath. However the linear velocity U of a rotating airfoil increases with increasing radius. Accordingly, the relative velocity V_r of the working medium at the airfoil leading edge increases with increasing radius, and at high enough rotational speeds, the airfoil experiences supersonic working medium flow velocities in the vicinity of its tip. Supersonic flow over an airfoil, while beneficial for maximizing the pressurization of the working medium, has the undesirable effect of reducing fan efficiency by introducing losses in the working medium's velocity and total pressure. Therefore, it is typical to sweep the airfoil's leading edge over at least a portion of the blade span so that the working medium velocity component in the chordwise direction (perpendicular to the leading edge) is subsonic. Since the relative velocity V_r increases with increasing radius, the sweep angle typically increases with increasing radius as well. As shown in FIG. 4, the sweep angle σ at any arbitrary radius is the acute angle between a line 54 tangent to the leading edge 28 of the airfoil 22 and a plane 56 perpendicular to the relative velocity vector V_r. The sweep angle is measured in plane 58 which contains both the relative velocity vector and the tangent line and is perpendicular to plane 56. In conformance with this definition sweep angles σ_1 and σ_2, referred to hereinafter and illustrated in FIGS. 2, 3 and 6 are shown as projections of the actual sweep angle onto the plane of the illustrations.

Sweeping the blade leading edge, while useful for minimizing the adverse effects of supersonic working medium velocity, has the undesirable side effect of creating an endwall reflection shock. The flow of the working medium over the blade suction surface generates pressure waves 60 (shown only in FIG. 1) which extend along the span of the blade and reflect off the case. The reflected waves 62 and the incident waves 60 coalesce in the vicinity of the case to form an endwall shock 64 across each interblade passage. The endwall shock extends radially inward a limited distance, d, from the case. As best seen in the prior art (phantom) illustration of FIG. 3, each endwall shock is also oblique to a plane 67 perpendicular to the rotational axis so that the shock extends axially and circumferentially. In principle, an endwall shock can extend across multiple interblade passages and affect the working medium entering those passages. In practice, expansion waves (as illustrated by the representative waves 68) propagate axially forward from each airfoil and weaken the endwall shock from the airfoil's leading neighbor so that each endwall shock usually affects only the passage where the endwall shock originated. In addition, the supersonic character of the flow causes passage shocks 66 to extend across the passages. The passage shocks, which are unrelated to endwall reflections, extend from the leading edge of each blade to the suction surface of the blade's leading neighbor. Thus, the working medium is subjected to the aerodynamic losses of multiple shocks with a corresponding degradation of engine efficiency.

The endwall shock can be eliminated by making the case wall perpendicular to the incident expansion waves so that the incident waves coincide with their reflections. However other design considerations, such as constraints on the flowpath area and limitations on the case construction, may make this option unattractive or unavailable. In circumstances where the endwall shock cannot be eliminated, it is desirable for the endwall shock to coincide with the passage shock since the aerodynamic penalty of coincident shocks is less than that of multiple individual shocks.

According to the present invention, coincidence of the endwall shock and the passage shock is achieved by uniquely shaping the airfoil so that the airfoil intercepts the endwall shock extending from the airfoil's leading neighbor and results in coincidence between the endwall shock and the passage shock.

A swept back airfoil according to the present invention has a leading edge 28, a trailing edge 30, a root 24 and a tip 26 located at a tip radius r_{tip}. An inner transition point 40 located at an inner transition radius r_t-inner is the axially forwardmost point on the leading edge. The leading edge of the airfoil is swept back by a radially varying first sweep angle σ_1 in an intermediate region 70 of the airfoil (in FIG. 2 plane 56 appears as the line defined by the plane's intersection with the plane of the illustration and in FIG. 3 the tangent line 54 appears as the point where the tangent line penetrates the plane of the Figure). The intermediate region 70 is the region radially bounded by the inner transition radius r_t-inner and the outer transition radius r_t-outer. The first sweep angle, as is customary in the art, is nondecreasing with increasing radius, i.e. the sweep angle increases, or at least does not decrease, with increasing radius.

The leading edge 28 of the airfoil is also swept back by a radially varying second sweep angle σ_2 in a tip region 74 of the airfoil. The tip region is radially bounded by the outer transition radius r_t-outer and a tip radius r_{tip}. The second sweep angle is nonincreasing (decreases, or at least does not increase) with increasing radius. This is in sharp contrast to the prior art airfoil 22' whose sweep angle increases with increasing radius radially outward of the inner transition radius.

The beneficial effect of the invention is appreciated primarily by reference to FIG. 3 which compares the invention (and the associated endwall and passage shocks) to a prior art blade (and its associated shocks) shown in phantom. Referring first to the prior art illustration in phantom, the endwall shock 64 originates as a result of the pressure waves 60 (FIG. 1) extending along the suction surface of each blade. Each endwall shock is oblique to a plane 67 perpendicular to the rotational axis, and extends across the interblade passage of origin. The passage shock 66 also extends across the flow passage from the leading edge of a blade to the suction surface of the blade's leading neighbor. The working medium entering the passages is therefore adversely influenced by multiple shocks. By contrast, the nonincreasing character of the second sweep angle of a swept back airfoil 22 according to the invention causes a portion of the airfoil leading edge to be far enough forward (upstream) in the working medium flow that the section of the airfoil radially coextensive with the endwall shock extending from the airfoil's leading neighbor intercepts the endwall shock 64 (the unique sweep of the airfoil does not appreciably affect the location or orientation of the endwall shock; the phantom endwall shock associated with the prior art blade is illustrated slightly upstream of the endwall shock for the airfoil of the invention for illustrative clarity). In addition, the passage shock 66 (which remains attached to the airfoil leading edge and therefore is translated forward along with the leading edge) is brought into coincidence with the endwall shock so that the working medium does not encounter multiple shocks.

The embodiment of FIGS. 2 and 3 illustrates a blade whose leading edge, in comparison to the leading edge of a conventional blade, has been translated axially forward parallel to the rotational axis (the corresponding translation of the trailing edge is an illustrative convenience—the

location of the trailing edge is not embraced by the invention). However the invention contemplates any blade whose airfoil intercepts the endwall shock to bring the passage shock into coincidence with the endwall shock. For example, FIG. 5 illustrates an embodiment where a section of the tip region is displaced circumferentially (relative to the prior art blade) so that the blade intercepts the endwall shock **64** and brings it into coincidence with the passage shock **66**. As with the embodiment of FIG. 3, the displaced section extends radially inward far enough to intercept the endwall shock over its entire radial extent and brings it into coincidence with the passage shock **66**. This embodiment functions as effectively as the embodiment of FIG. 3 in terms of bringing the passage shock into coincidence with the endwall shock. However it suffers from the disadvantage that the airfoil tip is curled in the direction of rotation R. In the event that the blade tip contacts the rubstrip **46** during engine operation, the curled blade tip will gouge rather than abrade the rubstrip necessitating its replacement. Other alternative embodiments may also suffer from this or other disadvantages.

The invention's beneficial effects also apply to a blade having a forward swept airfoil. Referring to FIG. 6 and 7, a forward swept airfoil **122** according to the present invention has a leading edge **128**, a trailing edge **130**, a root **124** and a tip **126** located at a tip radius r_{tip}. An inner transition point **140** located at an inner transition radius r_t-inner is the axially aftmost point on the leading edge. The leading edge of the airfoil is swept forward by a radially varying first sweep angle σ_1 in an intermediate region **70** of the airfoil. The intermediate region is radially bounded by the inner transition radius r_t-inner and the outer transition radius r_t-outer. The first sweep angle τ_1 is nondecreasing with increasing radius, i.e. the sweep angle increases, or at least does not decrease, with increasing radius.

The leading edge **128** of the airfoil is also swept forward by a radially varying second sweep angle σ_2 in a tip region **74** of the airfoil. The tip region is radially bounded by the outer transition radius r_t-outer and the tip radius r_{tip}. The second sweep angle is nonincreasing (decreases, or at least does not increase) with increasing radius. This is in sharp contrast to the prior art airfoil **122'** whose sweep angle increases with increasing radius radially outward of the inner transition radius.

In the forward swept embodiment of the invention, as in the swept back embodiment, the nonincreasing sweep angle σ_2 in the tip region **74** causes the endwall shock **64** to be coincident with the passage shock **66** for reducing the aerodynamic losses as discussed previously. This is in contrast to the prior art blade, shown in phantom where the endwall shock and the passage shock are distinct and therefore impose multiple aerodynamic losses on the working medium.

In the swept back embodiment of FIG. 2, the inner transition point is the axially forwardmost point on the leading edge. The leading edge is swept back at radii greater than the inner transition radius. The character of the leading edge sweep inward of the inner transition radius is not embraced by the invention. In the forward swept embodiment of FIG. 6, the inner transition point is the axially aftmost point on the leading edge. The leading edge is swept forward at radii greater than the inner transition radius. As with the swept back embodiment, the character of the leading edge sweep inward of the inner transition radius is not embraced by the invention. In both the forward swept and back swept embodiments, the inner transition point is illustrated as being radially outward of the airfoil root. However the invention also comprehends a blade whose inner transition point (axially forwardmost point for the swept back embodiment and axially aftmost point for the forward swept embodiment) is radially coincident with the leading edge of the root. This is shown, for example, by the dotted leading edge **28"** of FIG. 2.

The invention has been presented in the context of a fan blade for a gas turbine engine, however, the invention's applicability extends to any turbomachinery airfoil wherein flow passages between neighboring airfoils are subjected to multiple shocks.

We claim:

1. A turbomachinery blade for a turbine engine having a cascade of blades rotatable about a rotational axis so that each blade in the cascade has a leading neighbor and a trailing neighbor, and each blade cooperates with its neighbors to define flow passages for a working medium gas, the blade cascade being circumscribed by a case and under some operational conditions an endwall shock extends a limited distance radially inward from the case and also extends axially and circumferentially across the flow passages, and a passage shock also extends across the flow passages, the turbomachinery blade including an airfoil having a leading edge, a trailing edge, a root, a tip and an inner transition point located at an inner transition radius radially inward of the tip, the blade characterized in that at least a portion of the leading edge radially outward of the inner transition point is swept and a section of the airfoil radially coextensive with the endwall shock extending from the leading neighbor intercepts the endwall shock so that the endwall shock and the passage shock are coincident.

2. A turbomachinery blade for a turbine engine having a cascade of blades rotatable about a rotational axis so that each blade in the cascade has a leading neighbor and a trailing neighbor, and each blade cooperates with its neighbors to define flow passages for a working medium gas, the blade cascade being circumscribed by a case and under some operational conditions an endwall shock extends a limited distance radially inward from the case and also extends axially and circumferentially across the flow passages and a passage shock also extends across the flow passages, the turbomachinery blade including an airfoil having a leading edge, a trailing edge, a root, a tip located at a tip radius, an inner transition point located at an inner transition radius radially inward of the tip, and an outer transition point at an outer transition radius radially intermediate the inner transition radius and the tip radius, the blade having a tip region bounded by the outer transition radius and the tip radius, and an intermediate region bounded by the inner transition radius and the outer transition radius, the blade characterized in that the leading edge is swept in the intermediate region at a first sweep angle which is generally nondecreasing with increasing radius, and the leading edge is swept over at least a portion of the tip region at a second sweep angle which is generally nonincreasing with increasing radius so that the section of the airfoil radially coextensive with the endwall shock extending from the leading neighbor intercepts the endwall shock so that the endwall shock and the passage shock are coincident.

3. The turbomachinery blade of claim 1 or 2 characterized in that the inner transition radius is coincident with the root at the leading edge of the blade.

* * * * *

4

INFORMATION DISCLOSURE CITATION

(Sheet 1 of 2)

Atty. Docket No.: 3600.100 Cont.	Application No.: 09/874,931
Applicants: David A. Spear et al.	
Filing Date: June 5, 2001	Group: 3745

U.S. PATENT DOCUMENTS

*Examiner Initial		Document Number	Date	Name	Class	Subclass	Filing Date (if appropriate)
		1964525	6/26/34	McMahan	170	159	
		2154313	4/11/39	McMahan	230	274	
		2628768	2/17/53	Kantrowitz			
		2660401	11/24/53	Hull, Jr.	416	243	
		2689681	9/21/54	Sabatiuk			
		2915238	12/1/59	Szydlowski	230	134	
		2934259	4/26/60	Hausmann	415	181	
		2935246	5/3/60	Roy	415	181	
		3416725	12/17/68	Bohanon	230	259	
		3444817	5/20/69	Caldwell	103	88	
		3692425	9/19/72	Erwin	415	181	
		3989406	11/2/76	Bliss	415	1	
		4012165	3/15/77	Kraig	415	145	
		4012172	3/15/77	Schwaar et al.	416	228	
		4274810	6/23/81	Nishikawa	416	186R	

FOREIGN PATENT DOCUMENTS

		Document Number	Date	Country	Class	Subclass	Translation (Yes/No)
	FR	2459387	1/9/81	France			Yes
	EP	0266298	5/4/88	Europe			
	SU	1528965	12/15/89	U.S.S.R.			No

OTHER DOCUMENTS (Including Author, Title, Date, Pertinent Page, Etc.)

		Puterbaugh et al., "Design of a Rotor Incorporating Meridional Sweep and Circumferential Lean for Shock Loss Attenuation," February 1987, Contract AFWAL-TR-86-2013, Aero Propulsion Laboratories, Air Force Wright Aeronautical Laboratories, Wright-Paterson Air Force Base, Ohio
		Cheatham et al., "Parametric Blade Study," November 1989, Report No. WRDC-TR-89-2121, Aero Propulsion and Power Laboratory, Wright Aeronautical Research & Development Center, Wright-Patterson Air Force Base, Ohio
		European Search Report, dated February 25, 1998, in EP 774,567

EXAMINER	DATE CONSIDERED

*Examiner: Initial if reference considered, whether or not citation is in conformance with MPEP 609. Draw line through citation if not in conformance and not considered. Include copy of this form with next communication.

Form PTO-1449

INFORMATION DISCLOSURE CITATION (Sheet 2 of 2)	Atty. Docket No.: 3600.100 Cont.	Application No.: 09/874,931
	Applicants: David A. Spear et al.	
	Filing Date: June 5, 2001	Group: 3745

U.S. PATENT DOCUMENTS

*Examiner Initial		Document Number	Date	Name	Class	Subclass	Filing Date (if appropriate)
		4358246	11/9/82	Hanson et al.	416	223R	
		4370097	1/25/83	Hanson et al.	416	228	
		4408957	10/11/83	Kurzrock et al.	415	181	
		4714407	12/22/87	Cox et al.	415	192	
		4726737	2/23/88	Weingold et al.	416	223A	
		4737077	4/12/88	Vera	416	242	
		4784575	11/15/88	Nelson et al.	416	226	
		5064345	11/12/91	Kimball	416	169A	
		5112192	5/12/92	Weetman	416	201A	
		5167489	12/1/92	Wadia et al.	415	182.1	
		5408826	4/25/95	Stewart et al.	60	226.1	
		5584661	12/17/96	Brooks	416	238	
		6071077	06/06/00	Rowlands	416	223A	

FOREIGN PATENT DOCUMENTS

		Document Number	Date	Country	Class	Subclass	Translation (Yes/No)
	WO	9107593	5/30/91	PCT			
	EP	0774567	5/21/97	Europe			
	EP	0801230	10/15/97	Europe			

OTHER DOCUMENTS (Including Author, Title, Date, Pertinent Page, Etc.)

		European Patent Office Official Action, dated September 24, 1998, in EP 774,567
		European Patent Office Official Action, dated January 21, 2002, in EP 801,203
		Leading edge sweep angle profiles of fan blades of Pratt & Whitney PW305 and PW306 gas turbine engines

EXAMINER	DATE CONSIDERED

*Examiner: Initial if reference considered, whether or not citation is in conformance with MPEP 609. Draw line through citation if not in conformance and not considered. Include copy of this form with next communication.

Form PTO-1449

US006071077A

United States Patent [19]

Rowlands

[11] Patent Number: 6,071,077

[45] Date of Patent: Jun. 6, 2000

[54] SWEPT FAN BLADE

[75] Inventor: **Paul A. Rowlands**, Bristol, United Kingdom

[73] Assignee: **Rolls-Royce PLC**, London, United Kingdom

[21] Appl. No.: **09/168,968**

[22] Filed: **Oct. 9, 1998**

Related U.S. Application Data

[63] Continuation-in-part of application No. 08/819,269, Mar. 18, 1997, abandoned.

[30] **Foreign Application Priority Data**

Apr. 9, 1996 [GB] United Kingdom 9607316

[51] **Int. Cl.**^{7} **F04D 29/34**

[52] **U.S. Cl.** .... **416/223 A**; 416/DIG. 2; 416/DIG. 5

[58] **Field of Search** .... 416/223 A, DIG. 5, 416/DIG. 2

[56] **References Cited**

U.S. PATENT DOCUMENTS

3,989,406	11/1976	Bliss .
4,358,246	11/1982	Hanson et al. .
4,488,399	12/1984	Robey et al. .
4,726,737	2/1988	Weingold et al. .
4,790,724	12/1988	Bousquet et al. .
5,064,345	11/1991	Kimball .
5,249,922	10/1993	Sato et al. .
5,642,985	7/1997	Spear et al. .

FOREIGN PATENT DOCUMENTS

0265335	4/1988	European Pat. Off. .
0 266 298	5/1988	European Pat. Off. .
0 385 913	9/1990	European Pat. Off. .
2-104-975	3/1983	United Kingdom .
2-170-868	8/1986	United Kingdom .
2-197-913	6/1988	United Kingdom .
WO 81/00243	2/1981	WIPO .

Primary Examiner—John Kwon
Attorney, Agent, or Firm—Oliff & Berridge, PLC

[57] **ABSTRACT**

A swept fan blade design for the low pressure compressor rotor or fan rotor stage of a ducted fan gas turbine engine has a leading edge swept forward near the hub up to a first radial height, then rearward up to second radial height and finally is swept forward again up to the blade tip. The aerodynamic effect is to produce a mid-height bias to the airflow enabling the tip region to be given increased twist and to possess increased resistance to foreign object damage. The design also provides a rear radial blade stacking axis to help reduce internal stresses due to centrifugal forces.

13 Claims, 7 Drawing Sheets

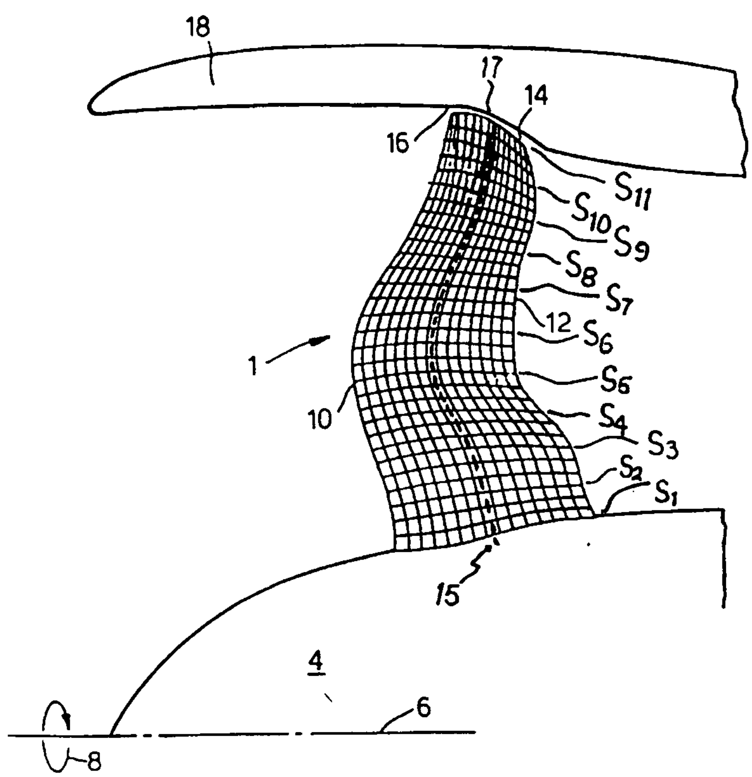


Fig.1.

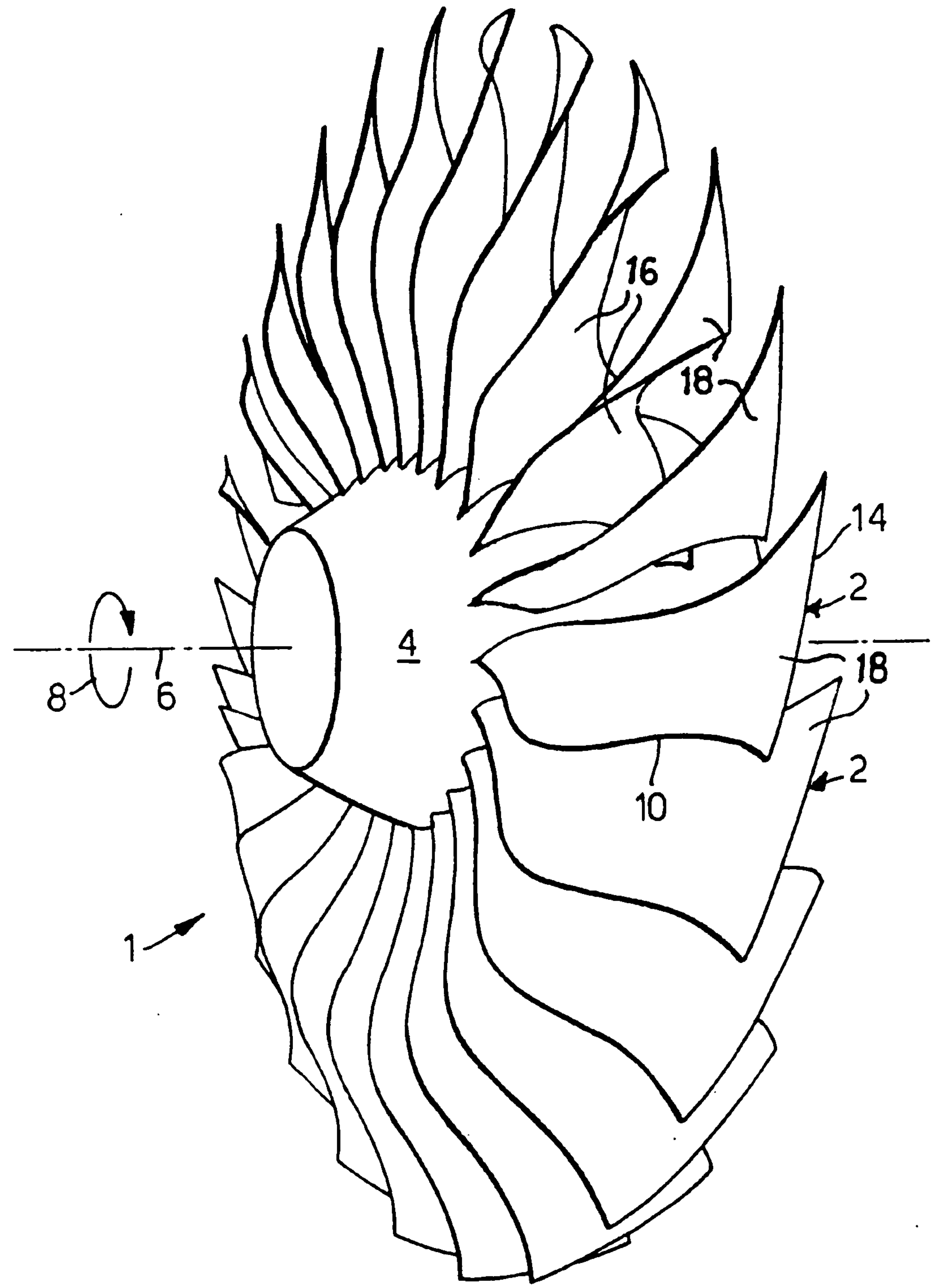


Fig.2a.

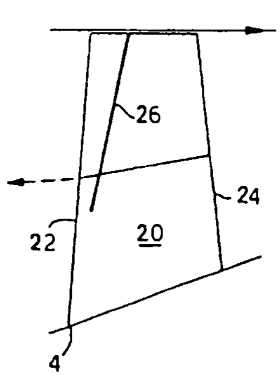


Fig.2b.

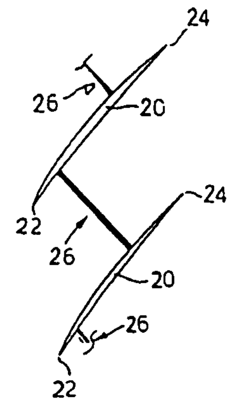


Fig.3a.

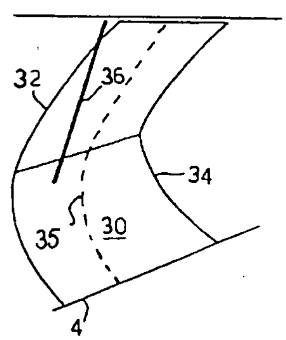


Fig.3b.

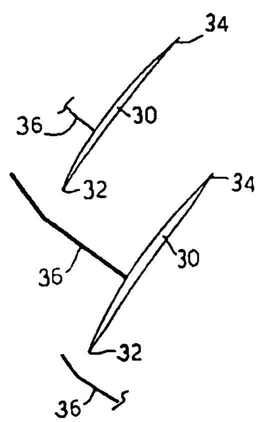


Fig.4a.

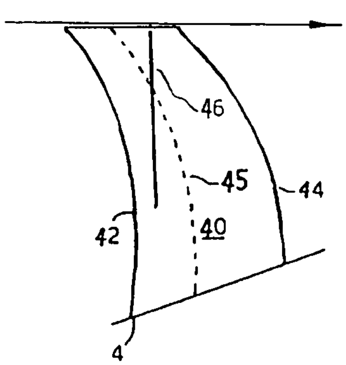


Fig.4b.

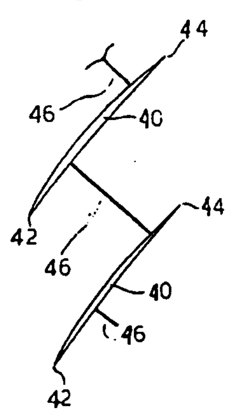


Fig.5a.

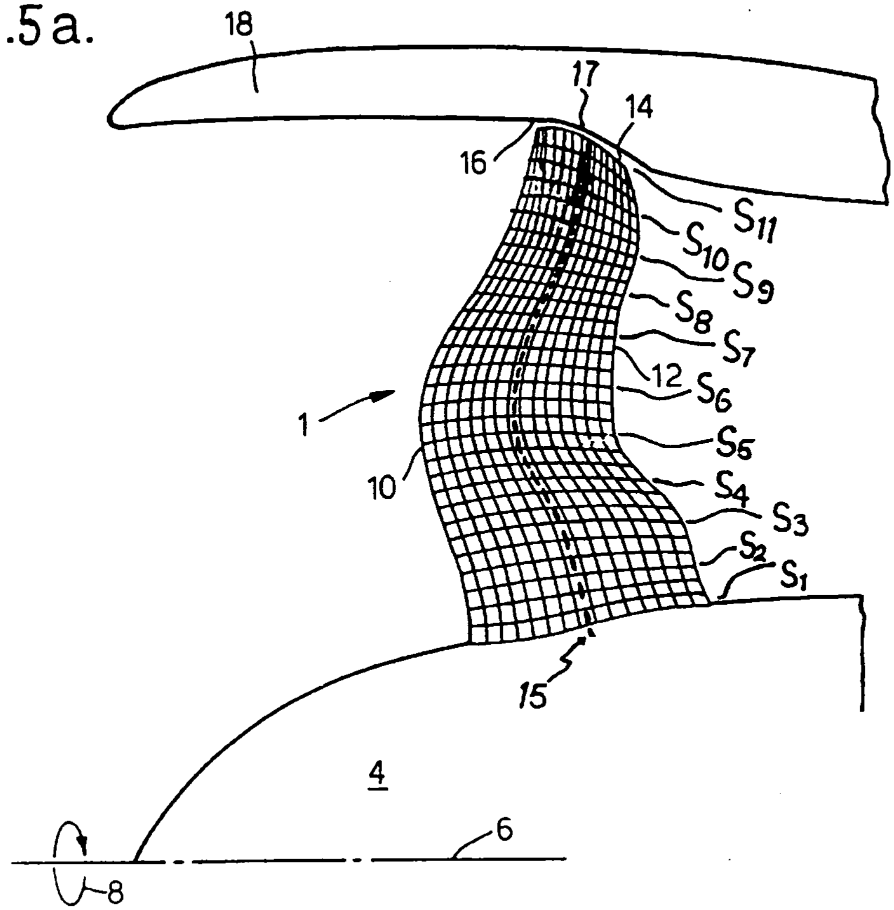


Fig.5b.

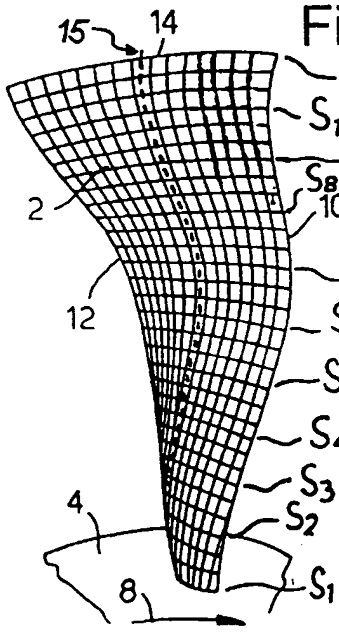


Fig.5c.

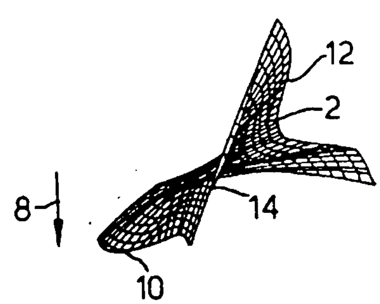


Fig.6.

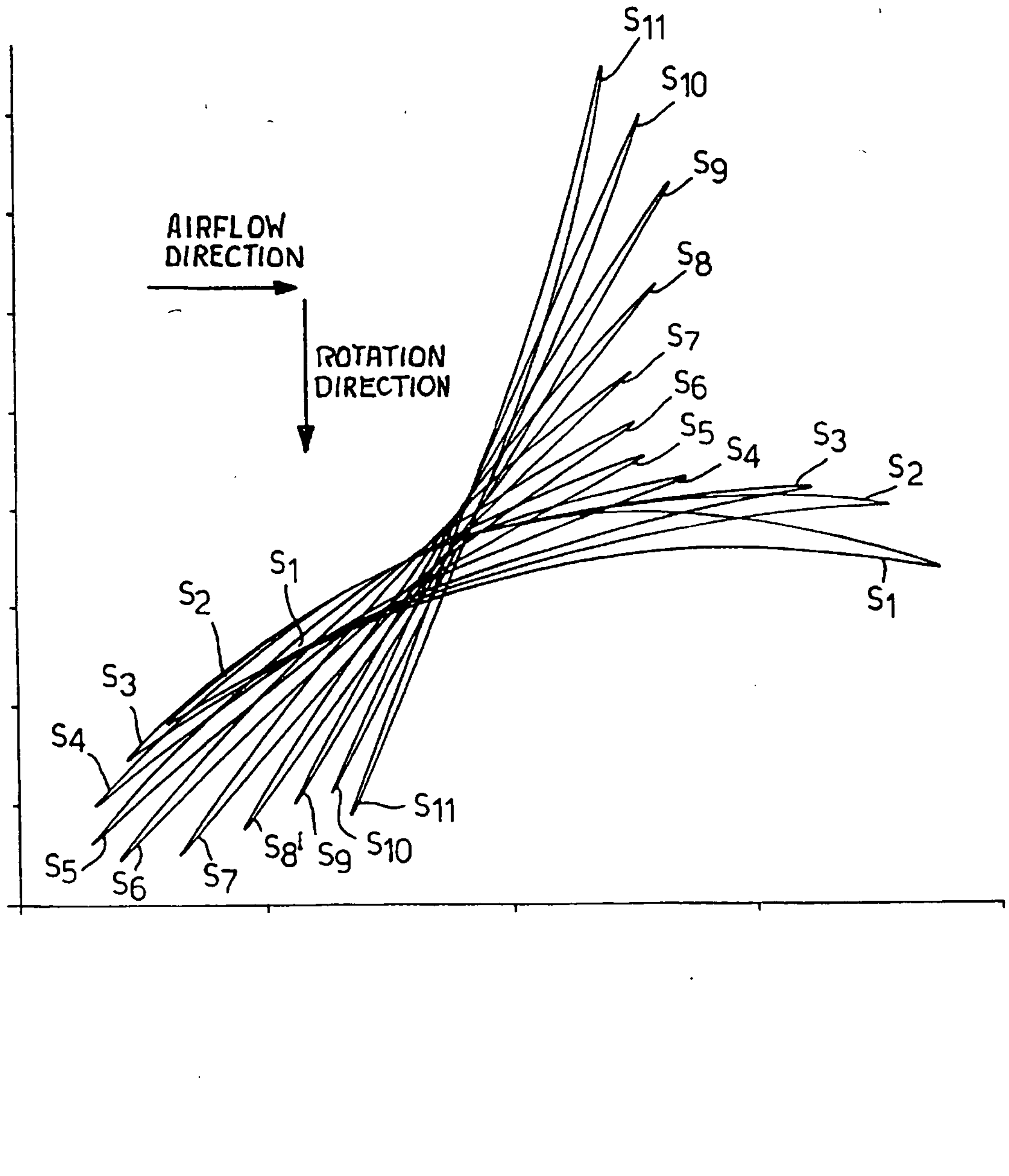

AIRFLOW DIRECTION
ROTATION DIRECTION
S11
S10
S9
S8
S7
S6
S5
S4
S3
S2
S1
S1
S2
S3
S4
S5
S6
S7
S8
S9
S10
S11

Fig.7a.

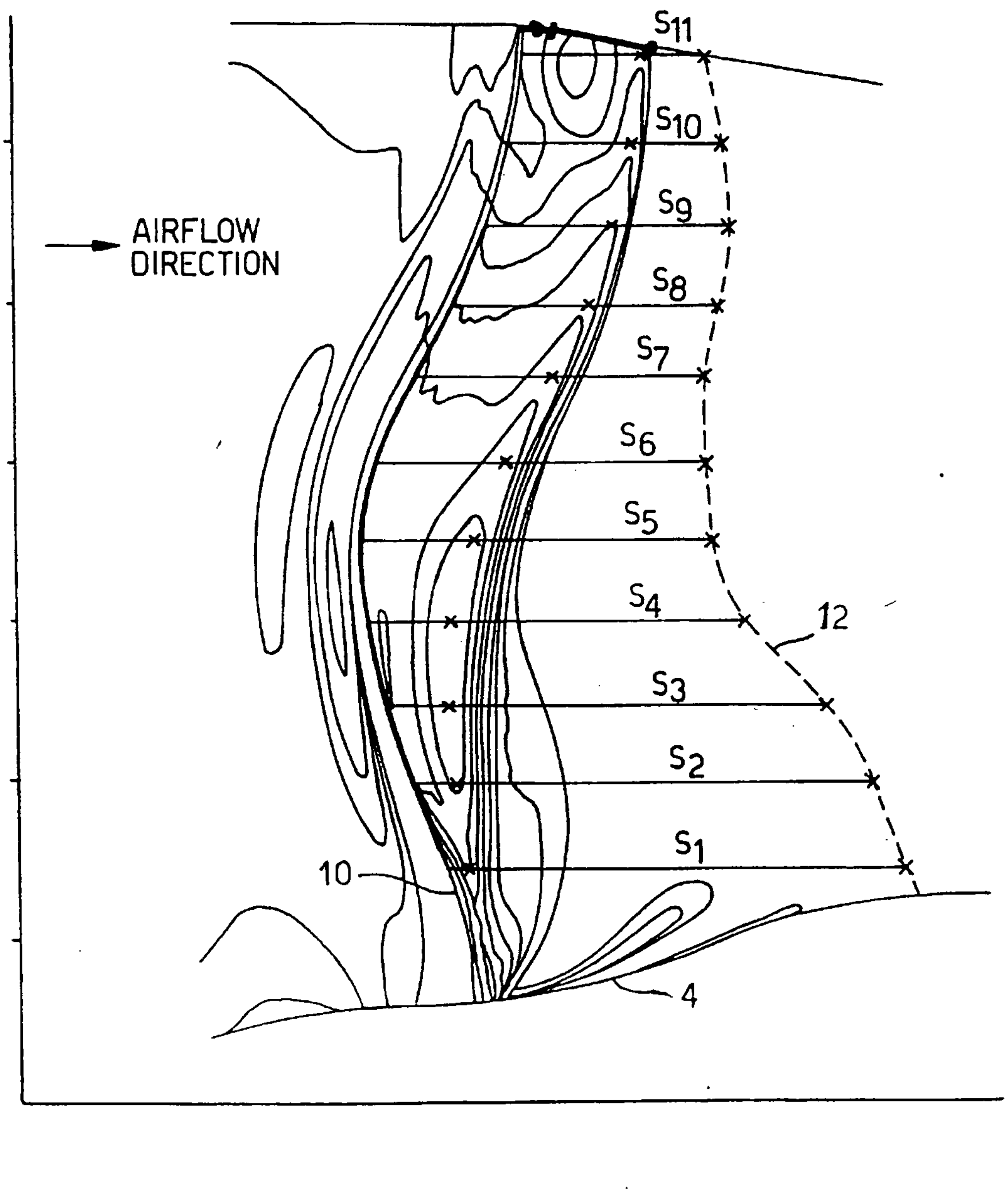


Fig.7b.

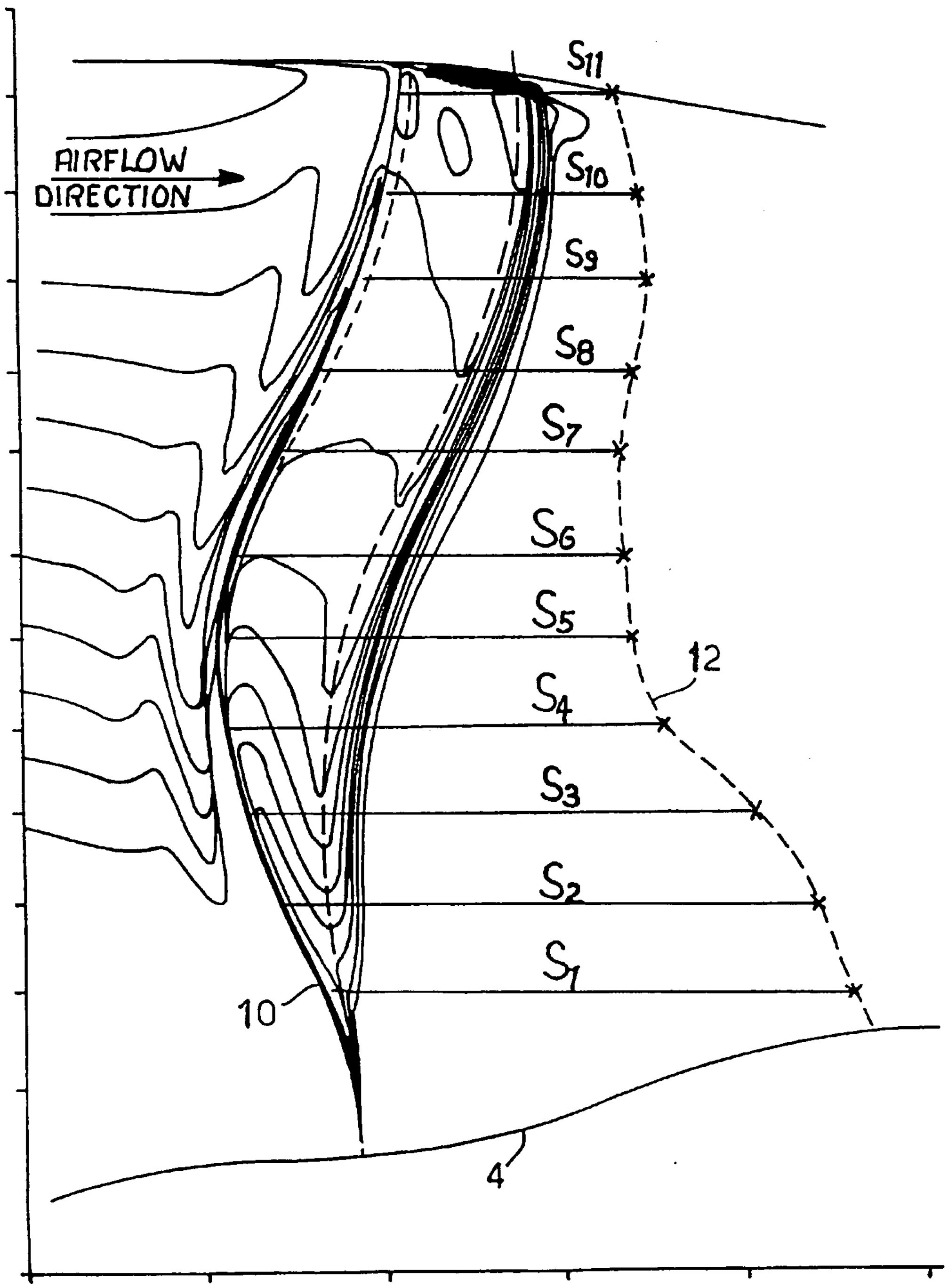


Fig.8.

PLANE SECTION NUMBER (n)	SUCTION SURFACE MINIMUM STATIC PRESSURE POINT			LEADING EDGE		
	SWEEP ANGLE (PS)	MACH ANGLE	MACH Nº	SWEEP ANGLE (LS)	MACH ANGLE	MACH Nº
1→2	0	66·6	1·09	−26·9	–	<1
2→3	−11·7	53·8	1·24	−25·4	–	<1
3→4	4·8	50·3	1·30	−21·6	74·1	1·04
4→5	15·1	47·8	1·35	1·4	62·2	1·13
5→6	26·9	46·9	1·37	6·4	57·2	1·19
6→7	34·9	46·0	1·39	19·1	53·8	1·24
7→8	34·9	44·4	1·43	25·5	49·8	1·31
8→9	25·4	43·2	1·46	23·2	46·9	1·37
9→10	26·9	42·5	1·48	10·9	44·4	1·43
10→11	27·6	41·1	1·52	−5·5	42·2	1·49

SWEPT FAN BLADE

This is a Continuation-in-Part of application Ser. No. 08/819,269 filed Mar. 18, 1997 now abandoned. The entire disclosure of the prior application is hereby incorporated by reference herein in its entirety.

The invention relates to a swept fan blade or compressor blade for a ducted fan gas turbine engine.

In particular the invention concerns the design of fan blades for a high bypass ratio engine of the kind used to power modem civil aircraft. A primary function of the fan is to generate propulsive thrust by adding energy to air passing through a fan duct increasing the pressure and momentum of the air. The performance of such a rotor is judged by the maximum thrust it produces, by way of maximum air flow and pressure rise, and the proportion of the energy input to the fan which is turned into useful thrust, characterised by the fan adiabatic efficiency. Fan stability is an important design consideration and a safety margin between the fan working line and its surge line is allowed to ensure stable operation. Therefore, it is standard practice to design a fan to achieve a given airflow and pressure rise at a chosen rotational speed which is below the maximum attainable to provide this stability margin.

In a high bypass ratio engine the fan is highly loaded and pumps a prodigious quantity of air at maximum speed. To achieve this a typical large engine of current design may have a fan of diameter up to 2 meters or more with individual blades of commensurate size. The fan blades provide thrust in the same way as lift is produced by an aircraft wing except that airspeed increases along the blade leading edge with increasing rotational radius. At some point along the leading edge the relative velocity becomes supersonic and generates a leading edge shock wave. Also as air is entrained in the passages between adjacent blades its pressure increases and at velocities greater than a critical number the pressure disturbances, unable to propagate upstream, produces a passage shock wave.

Shock waves represent a source of energy loss and because the fan produces so much of an engine's thrust any improvement in fan rotor efficiency has a significant effect on engine performance, particularly on fuel consumption.

The present invention has for one of its objectives to increase fan rotor efficiency above the levels currently achieved with existing designs and maintaining or improving the pressure rise with no erosion of the stability margin. It is desired to improve other rotor characteristics, for example resistance to foreign object damage at the same time as bringing about these other improvements.

According to the present invention there is provided a fan stage of a ducted fan gas turbine engine comprising a fan casing having an inner duct wall which in the region of a fan rotor is convergent in the downstream direction, a fan rotor comprising a multiplicity of swept fan blades spaced apart around a hub mounted concentrically with respect to the fan duct, each of said swept fan blades having a tip profile which in revolution conforms to the convergent duct wall, a leading edge of variable sweep angle which varies with increasing blade height or distance from the axis of rotation, said sweep angle having a forward sweep angle in a first height region between a root and a first intermediate radius, a rearward sweep angle in an intermediate height region between the first intermediate radius and a second intermediate radius, a forward sweep angle in a third height region between the second intermediate radius and the blade tip, a stagger angle which increases progressively with blade height.

Preferably the sweep angle of the leading edge is less than the complement of the angle of the Mach cone at any point on the leading edge. Usually sweep is employed to reduce the velocity of the airflow measured perpendicular to the leading edge to subsonic levels. This requires the sweep angle of the leading edge to be greater than the complement of the Mach cone angle. Sweep angle may be defined as the acute angle, at a point on the leading edge of a blade, between a tangent to the leading edge and a line perpendicular to the relative velocity vector measured in a common plane containing the tangent, the velocity vector and the line perpendicular thereto.

The invention and how it may be carried into practice will now be described in greater detail with reference to a particular embodiment illustrated in the accompanying drawings, in which:

FIG. **1** shows a perspective view of a swept blade fan rotor designed in accordance with the principles of the present invention,

FIGS. *2a*, *2b*; *3a*, *3b*; and *4a*, *4b* are side and tip views of a simple unswept blade, a rearward swept blade and a forward swept blade respectively illustrating the effect of sweep on the blade shock waves,

FIGS. *5a*, *5b* and *5c* show three orthogonal views of an individual blade from the rotor of FIG. **1**, on the surface of which a square grid pattern has been generated to illustrate the contours of the blade,

FIG. **6** is a view from a blade tip end of stacked aerofoil segments for a single blade from the rotor of FIG. **1** showing its change of chord length and stagger angle with stacking height,

FIGS. *7a* and *7b* show contour maps of static pressure on a blade suction surface and a corresponding contour map for relative Mach Nos. for the rotor blade of FIG. **1** , and

FIG. **8** is a table of sweep angle, Mach angle and Mach No at a number of points of successively greater blade height for the blade design of FIG. **1**.

Aerodynamic efficiency of the fan blades and thus of the fan stage, as discussed above, is a major fan attribute affecting engine performance and fuel consumption, but stability is also a major issue that must be addressed in any aerodynamic design. Any design must retain an adequate margin between its working line and the surge line on the pressure ratio against airflow surface. Surge is precipitated when a stall occurs in a compressor or fan cell when a sufficient portion of the passage shock front is expelled. A design objective is thus to have the passage shock front located sufficiently far downstream at the design point to protect the stall margin. Almost inevitably stability and efficiency are counteractive and stability may be bought at the expense of efficiency by over-compromise resulting in lost efficiency while designing for the maximum flow of air leads to instability.

However, blade design is not a straightforward trade-off between the factors affecting stability and efficiency and other variable features and their consequences contribute to the final form of a particular design.

The generation and behaviour of shock waves in rotors made of the basic different types of blade having no sweep, rearward sweep, and forward sweep is illustrated simply in FIGS. *2a*,*2b*; *3a*,*3b* and *4a*,*4b* of the accompanying drawings. To which further reference will now be made by way of summary of the basic aerodynamic effects of the present concern.

FIGS. *2a* and *2b* show side and tip views of a conventional, unswept rotor blade **20**, i.e. a straight edged blade, in these simple diagrams the twist and lean of the blades is omitted for clarity. In this type of design the aerofoil sections over most of the blade height operate in a

transonic flow regime, that is, the relative velocity of the air flow with respect to the rotor is supersonic when it passes over the leading edge 22 of the blade but is decelerated often to subsonic speed before passing the trailing edge 24. Much of this deceleration takes place in a step pressure discontinuity or shock front 26 which extends as a two-dimensional surface across a flow channel between adjacent blades, and therefore is bounded on one side by the suction surface of one blade and the pressure surface of an adjacent rotor blade. This discontinuity 26, is known as a passage shock wave.

The passage shock wave adds energy to the air, some of which cannot be recovered as thrust, and this energy loss in the shock wave contributes significantly to the total inefficiency of the fan rotor. Shock wave theory states that these losses are reduced when the Mach number of the airflow immediately ahead of the shock wave, and measured perpendicular to the wave front, is reduced. Therefore rotor efficiency may be increased by leaning the shock wave so that the air flow meets the discontinuity at a more oblique angle. The shock wave lean is achieved by sweeping the leading edge of the blades. Two forms of sweep, forwards and reverse, and the shock fronts produced thereby are illustrated in the side and tip view of FIGS. 3*a*, 3*b* and 4*a*,4*b*.

A simple form of rearward swept rotor blade 30 is illustrated by FIGS. 3*a* and 3*b*, any twist and lean of a typical practical blade is omitted for clarity and simplicity. In effect the stacked aerofoil segments of a conventional blade, i.e. compared to the blade of FIGS. 2*a*, are shifted axially rearward to provide a swept leading edge 32 to provide a swept leading edge 32. Forward sweep is employed near the hub 4 to counteract the rearward sweep of the outboard sections of blade 30 in order to make the design mechanically feasible.

The stacking axis, shown by dashed line 35, thus follows a curved path with increasing blade height which in a radial plane curves forward near the hub 4 and then rearwards, effectively tracking the sweep of the leading edge 32. Clearly, therefore centrifugal forces acting at points along the stacking axis 35 produce force couples in the radial plane which generate longitudinal stresses in the blade 30.

Rearward swept blades producing an oblique shock front relative to air stream flow direction generally exhibit good efficiency levels but at the expense of stability as a result of the shock front converging with the blade leading edge at blade heights towards the tip. Blade integrity levels are reasonably good although the rearward curve of the locus of the centres of gravity and pressure again produces high stress levels.

A forward swept blade 40, as illustrated by FIG. 4*a* and 4*b*, tends to be very stable because the passage shock front 46 is positioned well back in the throat passage and generally its displacement behind the leading edge 42 increases with blade height. While the forward sweep does not necessarily introduce poor aerodynamic efficiency internal stress levels resulting from the forward curved locus 45 of the centres of gravity and pressure of stacked blade segments are very high. As a result the blades individually have to be stronger and usually thicker, which tends to increase weight and the integrity problems.

Also shown in FIGS. 4*a* and 4*b* is the position of the shock surface 46 near to the suction surface. At a given aerofoil segment at a specific distance from the engine centre line (radial height) the position of the shock wave 46 is a function not only of the shape and position of this segment, but also of the position of the shock wave on other segments i.e. at different radial heights. This means that when a swept blade is designed as described above, the two-dimensional surface of shock wave 46 tends to shift less than the shift of the aerofoil segments, i.e. the sweep of the shock surface is less than the sweep of the leading edge 42. This results in the shock wave being closer to or even in front of the leading edge 42 near the suction surface towards the tip of the blade, as shown in FIGS. 4*a* and 4*b*.

Surge margin or stability is another design criterion to be considered. At the normal operating pressure rise and air flow rate the rotor must have sufficient safety margin to avoid flutter or surge. This margin is eroded when the operating pressure rise is increased. In such circumstances the shock wave is caused to move forwards towards the leading edge of the rotor blades. Eventually a limiting point is reached when the rotor can no longer maintain stable operation and the rotor will either suffer a violent reversal of airflow, known as a surge, or heavy vibration caused by local oscillations in airflow, known as flutter.

The present invention, however, overcomes these inherent stability problems of sweep by employing a number of specific novel physical features to move a shock surface rearwards, away from the leading edge, at the blade tip.

Mechanical loads and distribution caused by the high centrifugal forces produced by high speed rotor rotations have to be taken into account also, and can affect the integrity of blade designs. Since the relative velocity between blade and air stream increases with blade height, as well as the rotational speed, the aerodynamic design changes according to the conditions at the design point and this appears as, in particular, increasing blade twist in addition to blade sweep. Taken together, therefore, any blade design possesses a characteristic twist or stagger which varies with blade height; and a lean in which notional blade chordal sections are relatively displaced with blade height to provide forward or rearward sweep, as well possibly as circumferential lean. Thus, the locus of the centres of gravity of successive chordal cross-sections at increasing blade height normally deviates significantly from a radial path and as a result high internal stress levels can be generated. In order to contain these stresses blade thickness may have to be increased, adding mass and tending to exacerbate the problems and possibly detracting from the aerodynamic efficiency of the design.

Tolerance to foreign object damage and in particular bird strike, is another important design consideration. Resulting damage is caused by the magnitude of impact energy and the amount imparted to the blade which is measured in a direction normal to the blade surface.

A preferred embodiment of the present invention will now be described with reference to the accompanying drawings. Referring to the perspective view of a fan rotor in FIG. 1 although this is a view of a fan rotor incorporating swept blades in accordance with the present invention. There is shown a circular array of identical fan blades, of which one is referenced at 2, equidistantly spaced apart around the periphery of a rotatable hub 4. The manner by which the blades 2 are mounted on the hub 4 has no significance in the present context, it is irrelevant whether the blades are demountable or permanently fixed to the hub or disc 4, or formed integrally therewith.

No general limitation is intended by, nor should be inferred from, this description of an exemplary embodiment. Similarly, the composition and construction of the blades 2 is outside the scope of this invention, that is whether the blades are solid or hollow, metal or composite, or monolithic or fabricated.

The hub 4 is rotatable about an axis 6 in the direction of arrow 8, and this direction of rotation defines for each blade

2, its leading edge 10 and trailing edge 12, and the pressure and suction surfaces 16,18 respectively. The blades are shroudless and each therefore has a plain, unshrouded tip 14. The stacking axis of the blade is generally indicated by the broken line 15. Thus the overall appearance of a fan rotor incorporating blades designed according to the present invention is illustrated in the perspective view of FIG. 1.

FIGS. 5a, 5b and 5c show three mutually orthogonal views of one of the blades in which a rectangular grid has been superimposed on the surfaces of the blade in order to highlight its surface contours. Particularly noticeable are the two changes of sign of the sweep angle of the leading edge; the way in which the trailing edge and the stacking axis 15 of the blade follow the variations and sign changes of the leading edge; and the increasing twist of the blade with increasing radial height.

FIG. 6 shows a view of the blade on a radial axis of eleven blade segment profiles references S_1–S_{11} taken at equidistantly spaced radial heights from blade root to tip which are superimposed or stacked one on another to illustrate the sweep, lean and twist of the blade at successively greater radial heights.

Referring to FIGS. 5a, 5b, 5c and FIG. 6 the rotor blade leading edge 10 from the root radially outwards with increasing radial height is swept forward from the hub 4 or root segment S_1 (FIG. 6) to a maximum forward segment S_5 at approximately mid-height from where the leading edge is swept rearwards through segments S_6 to S_{10}. At about 75% of radial height the increasing rearward sweep is begun to be blended out until at around segment S_{10} there is a change of sign in the inclination of the leading edge 10. Near the blade tip 14 the inclination of the leading edge changes to forward sweep in the segment S_{11}. Also between segments S_{10} and S_{11} the chord length of the aerofoil section is increased. The axial and lateral disposition, i.e. stacking, of the blade aerofoil segments indicated by the movement of the stacking axis 15 in FIG. 5 is also apparent from FIG. 6 although the axis has been omitted for clarity. The blade tip segment S_{11} is positioned forward of the blade hub segment S_1, such that the point halfway between the leading edge and the trailing edge of the tip segment is axially upstream of a corresponding point of the hub segment. It will be apparent that, although there is considerable movement of the leading and trailing edges, there is relatively little axial and lateral (circumferential) movement of the blade segment stacking axis over its radial height. The internal stress levels arising from centrifugal forces due to rotation are dependent upon the separation distance between the stacking axis 15 and a true radius about the rotors axis of rotation. Thus, the more nearly radial is the stacking axis the lower are the stress levels generated by rotation.

The stagger angle, or twist, of the blade segments and how it increases with radial height between segments also will be apparent from FIG. 6. In comparison with a known, conventional fan rotor of similar dimensions the stagger angle of the mid height segments (roughly from S_7 to S_8) is reduced and the stagger of the tip segments (from S_8 to S_{11}) increased. In the particular blade illustrated the stagger of the mid-height section is in the range approximately 30° to approximately 55° relative to the airflow direction, while the stagger of the tip region increases from the approximately 55° angle to almost 70° also relative to the airflow direction at the tip. This feature produces a distinct mid-height bias to the airflow distribution across the span of a blade, with the result that airflow through the mid-height regions of the flow passage is increased and the airflow through the tip regions is reduced. This has an advantageous effect on the overall efficiency of the blade. This distinctive airflow distribution is contained within a region relatively close to the front and rear faces of the rotor, that is within approximately one chord length upstream of the leading edge and a similar distance downstream of the trailing edge. The airflow in this region is deflected away from both the hub and the tip, and follows a curved trajectory towards the mid-height passage region. As a result airspeed of the air stream in these regions especially in the tip region close to the fan casing wall is reduced. This allows the stagger angle of the radially outermost aerofoil segments of the blade (roughly segments S_9 to S_{11}) to be increased, that is blade twist is significantly increased and the stacking axis is inclined forwards.

The inner surface of the annular fan casing in the region 17 immediately encircling the fan rotor is tapered in the downstream direction, that is the diameter of the region 17 is greater on the upstream side of the fan 1 compared to the downstream side. Generally it has the appearance of a short frusto-conical surface although to avoid aerodynamic turbulence abrupt contour changes are avoided and blended smoothly with the remaining inner surface of the casing. An angled casing of this kind has long been used by us to avoid complicated aerodynamic interference effects which might otherwise be brought about by reflection of the passage shock waves from the casing wall as has been described in U.S. Pat. No. 5,642,985. In the present invention the potential effect of reflected passage shock waves is lessened by the reduction of airspeed in the blade tip regions which naturally produces weaker shock waves. Thus, any proclivity to generation of a reflected endwall shock is further reduced by the effect of the blade design.

A further advantage accruing from the invention due to the increased twist of the tip region is its improved resistance to foreign object damage (FOD). As previously mentioned, in an FOD strike the energy imparted to, or absorbed by, is determined by the magnitude of the component of the relative velocity vector resolved in a direction perpendicular to the blade edge or blade surface at the point of impact. This component of the relative velocity vector is obtained resolving the result of a vector addition of the speed of the object in the direction of travel with the blade speed in the direction of rotation. It will be apparent upon consideration that the magnitude of this resolved component is reduced by a greater blade twist angle. The blade thickness is thus able to be reduced lowering the mass of the section of the blade. Consequently the centrifugal forces acting on the blade outer region are reduced so the internal stress suffered by the blade is reduced. However, it is not a straightforward matter of simply introducing more blade twist because of the effect on blade aerodynamic efficiency. Too much twist would just reduce the efficiency of the blade to a point where it became too inefficient for practical use, and would cause stability to collapse over a substantial proportion of the blade height. An important factor in the success of the present blade design is the mid-height bias to the airflow distribution which facilitates the reduction of the speed of the airflow at high blade radii thereby permitting the increase of blade twist and the reversal of leading edge sweep angle, characteristics which give the blade design a distinctive appearance. Thus, a radially outer region of the blade including the tip region is particularly distinctive, most noticably it has a forward sweep angle, increased blade twist and forward lean of the stacking axis. Because of the smooth blended nature of the changes it is not possible to precisely define the inner radial limit of this region, although approximately it accounts for about 15% but may be up to about 20% of the radially outermost portion of the blade. In this blade tip region the

sweep of the leading edge changes from rearward sweep to forward sweep. Furthermore, as a result of some of the improvements the blade tip region is less susceptible to foreign object damage, as explained below. Consequently, the blade tip region may be reduced in thickness. So it is a further characteristic of the blade's appearance that the tip region, when compared with a known blade designed for similar engine size and mass flow, is significantly thinner.

Because of the correlation between blade speed (i.e. linear speed as opposed to rotational speed) and radial distance from the axis of rotation impact energy also tends to increase with blade height. Thus, the greater angle of twist and any reduction of relative speed between an object and a blade that can be achieved serves to lessen impact energy and the severity of consequential damage. At blade heights towards the tip where relative speed is maximium the amount of energy transferred to a blade by impact with a foreign object in a direction normal to the blade surface is reduced by the increased angle of blade twist.

Further to the features described above the stability of the swept blade is maintained by employing only modest amounts of sweep, when compared to previous designs, which tend to set the sweep angle to be greater than the complement of the Mach cone angle at a given position on the aerofoil segment, usually the leading edges although other additional positions, e.g. minimum suction surface static pressure points, have been stipulated.

As mentioned above, in operation, each blade creates a shock wave front which at design speed is a predetermined distance behind the leading edge. Also on the suction surface of the blade the segment profiles combine to create a spanwise extending line of minimum static pressure points. FIG. 7*a* shows a plot of static pressure contours (isobars) on the suction surface of a fan blade according to the invention, and in the airflow regions immediately upstream of the leading edge **10** and downstream of the trailing edge **12**. The eleven segment profiles of FIG. 6 are indicated by horizontal lines S_1–S_{11}, that is the lines so referenced which are parallel to the airflow direction indicated by an arrow. Apparent are pressure gradients in the axial or airflow direction which reveal the spanwise distribution of the line of minimum static pressure points, in front of the shock wave on the blade surface marking the abrupt transition between subsonic and supersonic flow. The distribution of isobars also reveals the mid-height bias in the airflow over the blade surface and the tip region reduction.

FIG. 7*b* shows corresponding contours of relative Mach number for airflow over the blade suction surface. These clearly show the abrupt transitions at the blade leading edge and at the blade surface where the passage shock wave meets the surface. On the left hand, vertical axis the plane section numbers S_1–S_{11} are inscribed for easy cross-reference between drawings. The leading edge **10** of the blades is inscribed for ease of reference and by comparison of the position of the step in the pressure contours the relative location of the shock front on the blade surface can be observed. Again the reduction of airflow in the tip region adjacent the casing wall can also be discerned.

FIG. **8** contains a table showing sweep Angle, Mach angle, i.e. the angle of the Mach cone, and Mach No for both the line of minimum static pressure points and at the blade section profile segment S_1–S_{11}. The changes of the leading edge sweep angle will immediately apparent from the column headed "SWEEP ANGLE".

The swept blade design described above is presented as an exemplary embodiment of the invention. It will be appreciated that the described design is not the only swept blade design which may embody the invention defined in the following claims.

What is claimed is:

1. A fan stage of a ducted fan gas turbine engine, comprising

a fan casing having an inner duct wall which in a fan rotor region is convergent in the downstream direction; and

a fan rotor including a multiplicity of swept fan blades spaced apart around a hub mounted concentrically with respect to the fan duct, each of said swept fan blades having a tip profile which in revolution is convergent so as to substantially correspond to the convergent duct wall, a leading edge of variable sweep angle which varies with increasing blade height or distance from the axis of rotation, said sweep angle having a forward sweep angle in a first height region between the root and a first intermediate radius, a rearward sweep angle in an intermediate height region between the first intermediate radius and a second intermediate radius, a forward sweep angle in a third height region between the second intermediate radius and the tip of the blade, a stagger angle which increases progressively with blade height.

2. A fan stage of a ducted fan gas turbine engine as claimed in claim 1 wherein the blade has a tip region of up to about 20% of blade height characterised in that the stagger angle increases to approximately 70° at the tip relative to the airflow direction.

3. A fan stage of a ducted fan gas turbine engine as claimed in claim 2 wherein a blade tip region of up to about 20% of the height of the blade the sweep of the leading edge changes from rearward sweep to forward sweep.

4. A fan stage of a ducted fan gas turbine engine as claimed in claim **3** wherein the blade is further characterised in that the stagger angle of the mid-height region of the blade is in the range from approximately 30° to approximately 55° relative to the airflow direction.

5. A fan stage of a ducted fan gas turbine engine as claimed in claim 1 wherein the sweep angle of the leading edge of a swept fan blade at a point on the leading edge is less than the complement of the angle of a Mach cone at any other point on the leading edge of the blade at greater radius from the root.

6. A fan stage of a ducted fan gas turbine engine as claimed in claim **1** wherein the shape of the pressure surface of a swept fan blade and the suction surface thereof creates, in use, a line of minimum static pressure points on the suction surface of the blade, said line of minimum static pressure points is inclined with respect to the axial direction at a sweep angle which varies with span height of the blade, and has a negative value in a region of subsonic flow over the leading edge.

7. A fan stage of a ducted fan gas turbine engine as claimed in claim 6 wherein the sweep angle of the line of minimum pressure points at a point on the line is less than the complement of the angle of a Mach cone at any other point on the line.

8. A fan stage of a ducted fan gas turbine engine that is at least in part rotatable about an axis of rotation and defines a downstream direction along the axis of rotation, comprising:

a fan casing that defines an inner duct wall having a fan rotor region, the inner duct wall of the fan casing at the fan rotor region being convergent;

a hub disposed concentrically relative to the fan casing;

a fan rotor that includes multiple swept fan blades, the swept fan blades being spaced apart around the hub, each of the multiple swept fan blades having:

a tip profile that is convergent so as to substantially correspond to the convergent inner duct wall of the fan casing;

a leading edge that defines a variable sweep angle in a direction perpendicular to the axis of rotation, the leading edge including:

an inner region adjacent the hub, the inner region defining a forward sweep angle;

an intermediate region between the inner region and the fan casing, the intermediate region defining a rearward sweep angle; and

an outer region between the intermediate region and the fan casing, the outer region defining a forward sweep angle.

9. The fan stage according to claim **8**, wherein the intermediate region extends further than the inner region along the axis of rotation.

10. The fan stage according to claim **8**, wherein the inner duct wall of the fan casing at the fan rotor region is substantially convergent in the downstream direction.

11. The fan stage according to claim **8**, wherein the tip profile of the multiple swept fan blades are substantially convergent in the downstream direction.

12. The fan stage according to claim **8**, wherein the inner duct wall of the fan casing at the fan region is not parallel to the tip profile of each of the multiple swept fan blades.

13. The fan stage according to claim **8**, wherein each of the multiple swept fan blades includes a hub contacting surface that extends further than the tip profile along the axis of rotation.

* * * * *

6

United States Patent [19]

Schwaar et al.

[11] **4,012,172**

[45] **Mar. 15, 1977**

[54] **LOW NOISE BLADES FOR AXIAL FLOW COMPRESSORS**

[75] Inventors: **Pierre G. Schwaar**, Monroe; **John A. O'Connor**, Orange, both of Conn.

[73] Assignee: **Avco Corporation**, Stratford, Conn.

[22] Filed: **Sept. 10, 1975**

[21] Appl. No.: **612,220**

[52] **U.S. Cl.** .... **416/228; 416/223 A**

[51] **Int. Cl.**[2] **F04D 29/38**

[58] **Field of Search** .... 415/181, 119, DIG. 1, 415/79; 416/228, 223

[56] **References Cited**

UNITED STATES PATENTS

2,160,467	5/1939	Ward	416/228
2,359,466	10/1944	Currie	416/202
2,663,493	12/1953	Keast	416/228
2,839,239	6/1958	Stalker	415/181
2,841,325	7/1958	Weise	415/181 X
3,347,520	10/1967	Owczarek	416/228
3,467,197	9/1969	Spivey et al.	416/228
3,972,646	8/1976	Brown et al.	416/228

FOREIGN PATENTS OR APPLICATIONS

1,566	4/1926	Australia	416/228
378,677	7/1923	Germany	416/223
647,053	6/1937	Germany	416/223
1,903,642	8/1970	Germany	416/223
119,463	5/1919	United Kingdom	416/228
1,369,229	10/1974	United Kingdom	416/228

Primary Examiner—Everette A. Powell, Jr.
Attorney, Agent, or Firm—Irwin P. Garfinkle; Edmund S. Lee

[57] **ABSTRACT**

A rotor blade configuration is disclosed which greatly reduces noise generated by the low pressure compressor, or fan, of a turbofan engine. The leading edge of the blade is swept forwardly from its hub up to a point of sweep reversal and then swept rearwardly to the tip of the blade. The slope of the curved leading edge line relative to the direction of airflow is gradually decreased from the hub to the tip to maintain the velocity component, of air relative to the blade leading edge, subsonic, while the relative velocity of the air to the blade is above a critical value and supersonic. This substantially eliminates noise producing, standing shock waves along the blade leading edges. Also described are methods for determining the configuration of such blades to obtain these ends with a minimum increase in blade weight and a minimum blade bending and blade attachment stresses.

1 Claim, 8 Drawing Figures

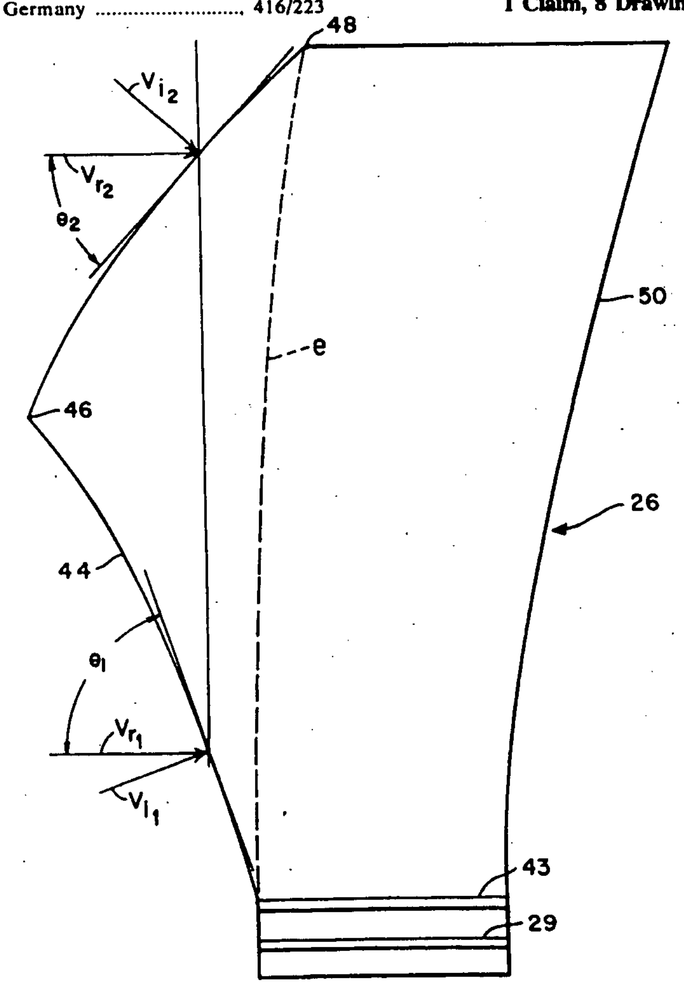


FIG 1

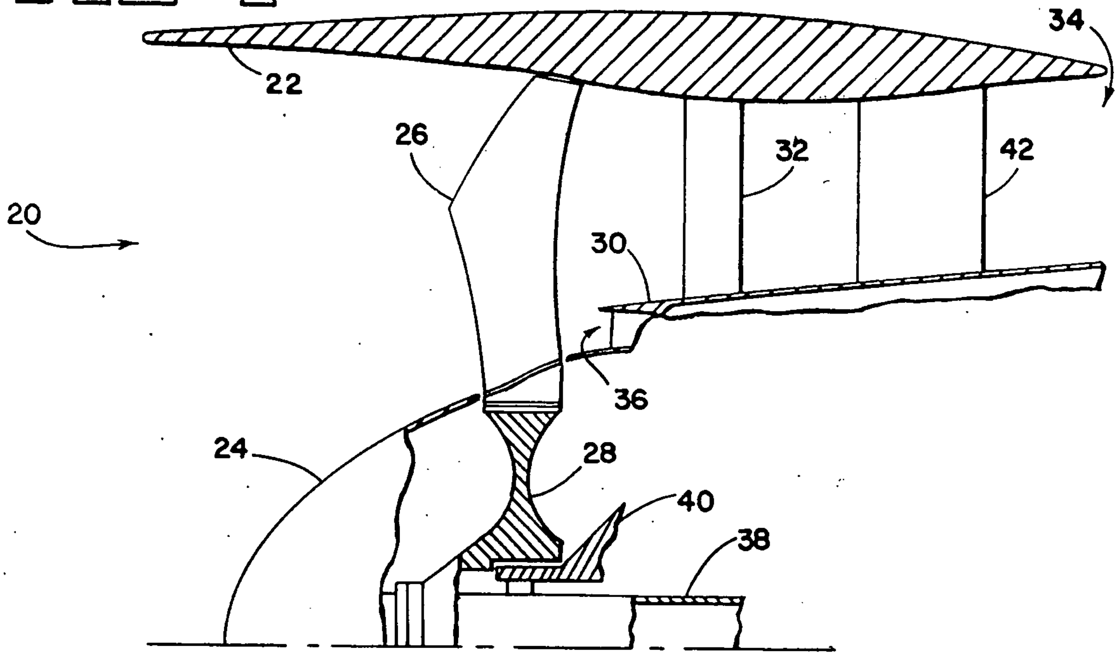


FIG 2

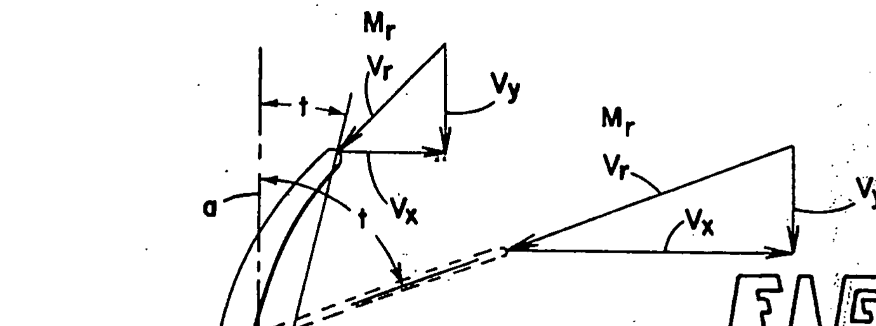


FIG 3

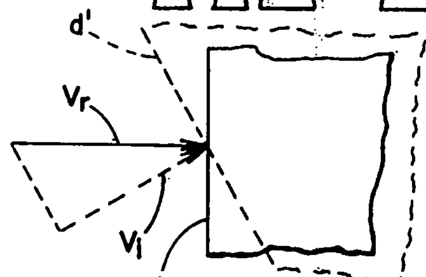


FIG 4

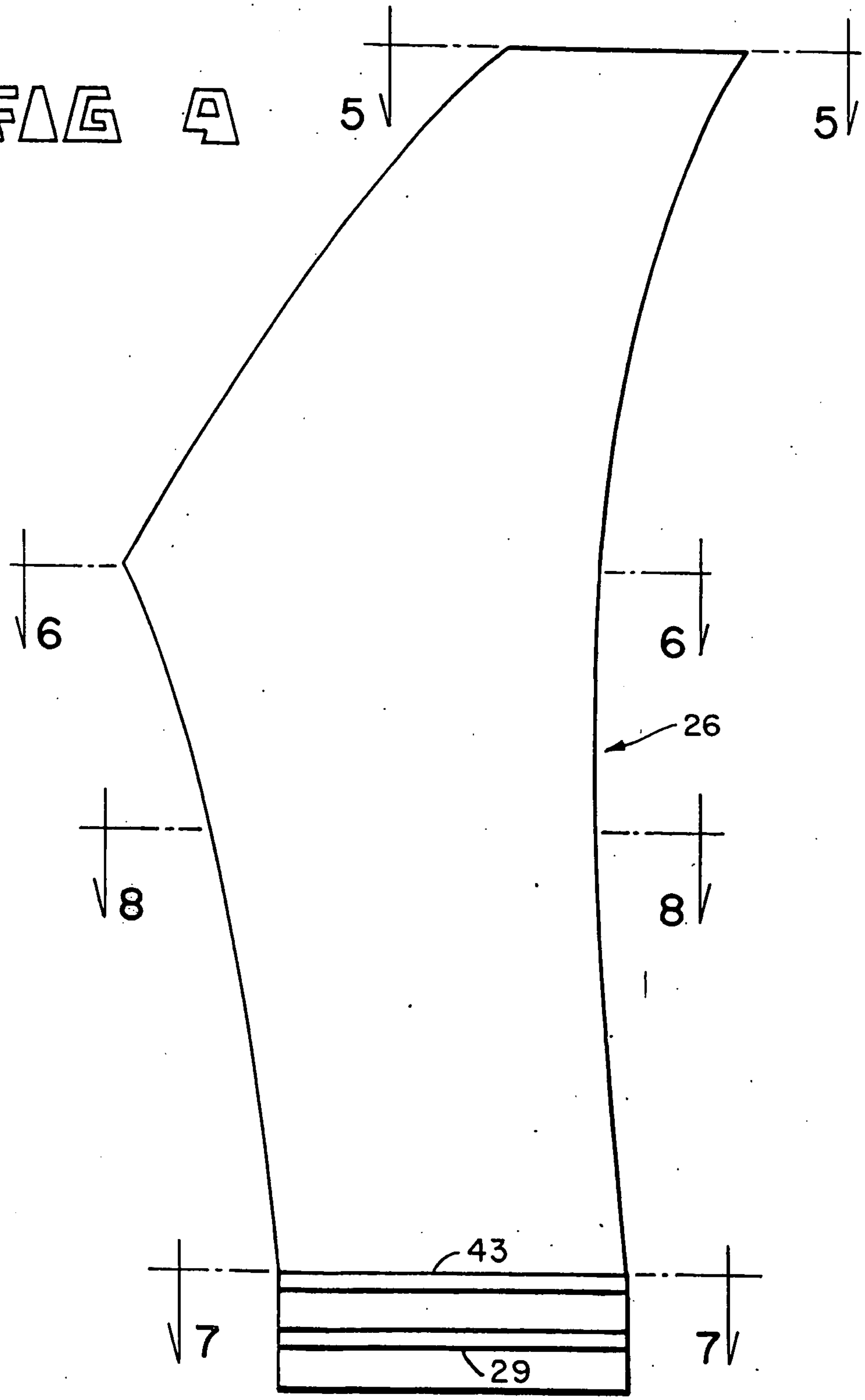


FIG 5

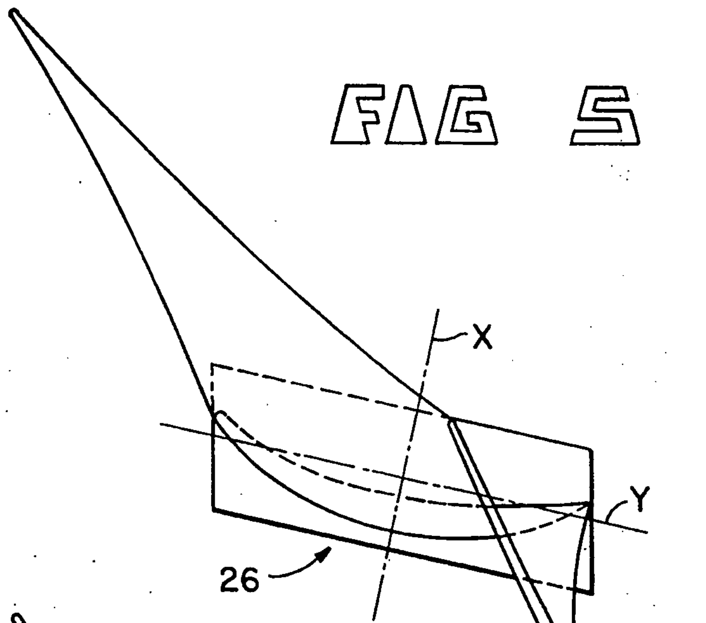


FIG 6

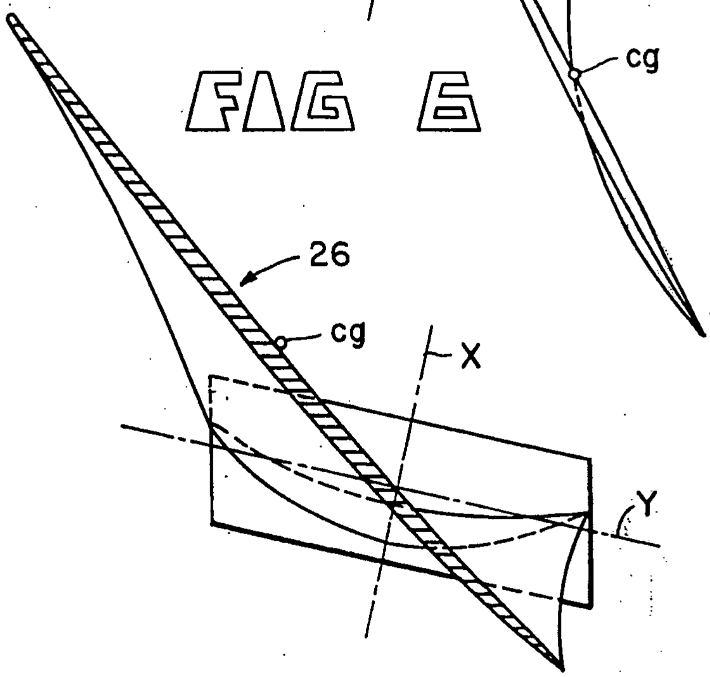


FIG 7

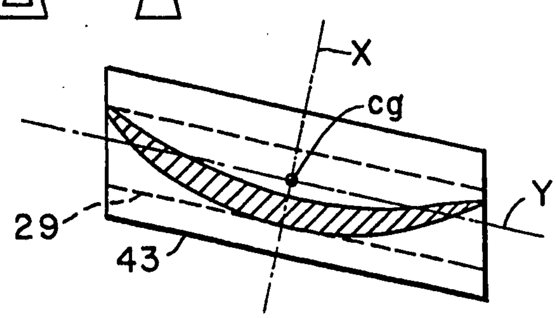
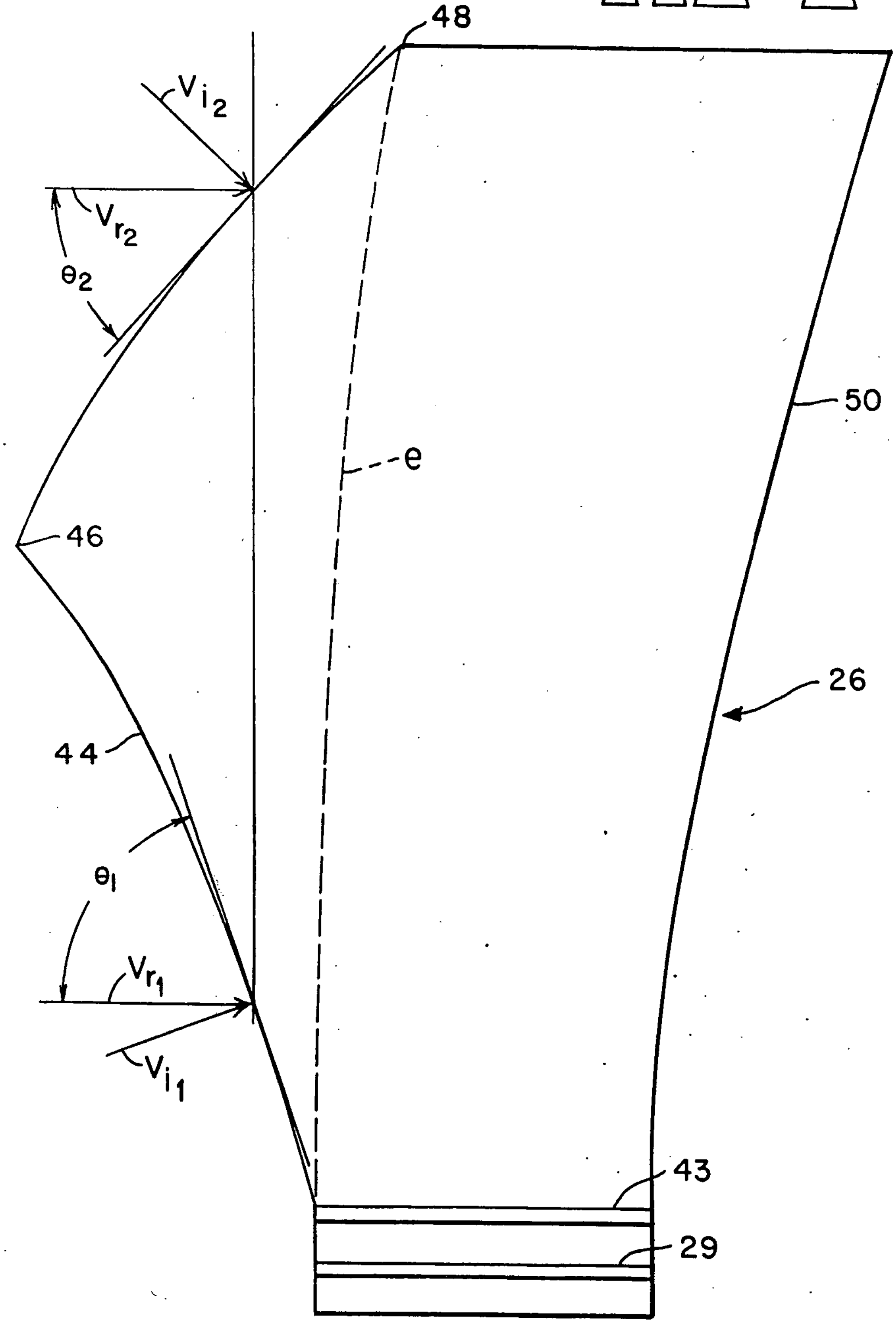


FIG. 8



LOW NOISE BLADES FOR AXIAL FLOW COMPRESSORS

The present invention relates to improvements in axial flow compressors, and particularly to noise abatement in low pressure compressors, known as fans, employed in the propulsion of aircraft.

Aircraft flight speeds have increased progressively with the advent of gas turbine propulsion systems. In early systems, the hot gas stream generated by such engines was discharged through a nozzle to provide the propulsive force. Later turbofan engines were developed which provided quieter operation and greater efficiency at high subsonic flight speeds. In a turbofan propulsion system, a portion of the gas stream energy is extracted to drive a low pressure compressor, commonly referenced as a fan, which pressurizes an annular air stream outwardly of the hot gas stream. This air stream, which bypasses the engine proper, is also discharged through a nozzle to supplement the propulsive force of the hot gas stream. Originally the ratio of bypass air to hot gas stream flow was relatively low, in the order of one to one. Such engines have now been superseded, for many applications, by engines with bypass ratios in the order of five to one and upwards to provide even greater efficiency.

Gas turbine engines inherently generate objectionable noise in discharging the hot gas stream through a propulsion nozzle. The introduction of turbofan engines was beneficial from a noise standpoint because of the reduction of the proportionate energy level of the hot gas stream discharged through a propulsion nozzle.

However, turbofan engines, and high bypass ratio engines in particular, created a new and different type of noise problem. In such engines the relative air flow velocity past the upper portion of the rotating fan blades is supersonic. This produces shock waves, along the leading edges of the fan blades, which generate objectionable, forward propagating noise referred to as "multiple pure tone noise." This noise frequently becomes an important part of the overall engine noise.

Shock waves may also be generated locally on the blade surface when the relative air flow velocity exceeds a critical, subsonic value. These shock waves do not generate the referenced "multiple pure tone" noise but are objectionable in that they cause undesirable energy losses. Critical velocity is a function of the sharpness of the leading edge and camber of the blade and other factors affecting local flow conditions. Generally, the Mach number of the critical velocity will range from 0.8 to slightly below 1.0.

It has previously been recognized that the generation of shock waves can be prevented by angling the leading edge of a fan compressor blade, or other airfoil relative to the direction of air flow. This is evidenced by the swept wings of high subsonic and supersonic flight speed aircraft and in specific reference to compressor blades by U.S. Pat. No. 2,633,493, granted to K. H. Keast. In both cases the swept leading edge causes the velocity component normal to the leading edge of the blade to be reduced and, dependent upon the slope of the leading edge relative to the direction of the airflow, such velocity component may be maintained below the critical velocity, or subsonic.

These teachings are not applicable, in a practical sense, to eliminating the generation of shock waves on the blades of high bypass ratio fans. This is due to the

high supersonic peripheral tip speeds needed to maintain high work input capacity at the hub section of the high tip-to-hub ratio rotors used in such fans. When these factors are taken into account, simply sweeping the leading edges of such blades, in the fashion done in the past, results in a blade which is either excessively heavy and/or subject to excessive stresses. Thus, while the known principle of sweeping the leading edges of compressor or fan blades is applicable to eliminating forward propagating, standing shock waves and the objectionable noise generated thereby, present teachings do not provide a fan blade practical for incorporation in advanced turbofan engines.

The principal object of the present invention is to reduce the noise generated by shock waves along the leading edges of axial flow compressor blades, particularly in low pressure compressors, known as fans, employed in high bypass ratio turbofan engines.

Another object of the present invention is to minimize, if not eliminate, energy losses due to the generation of localized shock waves where the relative flow velocity exceeds a critical value.

Another object of the present invention is to attain such ends with an improved, structurally sound blade having a minimum weight increase over a conventional blade.

Yet another object of the present invention is to provide improved methods for establishing the configuration of a compressor blade which fulfills the above ends.

These ends are obtained in an axial flow compressor comprising a rotor having a circumferential row of radial blades. Each blade is in the form of an airfoil having varying camber and setting angles from its hub end to its tip end.

The rate of rotation of the rotor, at a design speed, causes the relative velocity of air flow past at least a portion of the blade to exceed a sonic value above which shock waves could be generated along the leading edge of the blade if it were normal to the air flow direction.

In accordance with the invention, the leading edge, at least outwardly from the point where the relative velocity reaches the sonic value, is swept forwardly on a curve having a progressively decreasing slope relative to the direction of air flow. The forwardly swept curve continues to a point located inwardly from the tip end of the blade. From this point the leading edge of the blade is swept rearwardly to the tip end of the blade along a curve also having a progressively decreasing slope relative to the direction of air flow. The slope angle, at all points along said curved leading edge, is such that the velocity component of the incident air flow normal to the leading edge is maintained below the sonic velocity.

Preferably, the trailing edge of the blade is smoothly curved from its hub end to its tip end. Other preferred features are found in having conventional hub and tip end profiles and in having the blade profiles stacked so that the sum of the bending moments produced by the centrifugal and the aerodynamic blade forces is minimized.

The improved method of the invention is found in the steps of establishing the slope of the blade leading edge which maintains the air velocity component normal thereto below the sonic or the critical value at the desired rate of blade rotation, optimizing the location

of the point of sweep reversal and stacking the blade profiles so as to minimize the blade bending stresses.

The above and other related objects and features of the invention will be apparent from a reading of the following description of the disclosure, with reference to the accompanying drawings and the novelty thereof pointed out in the appended claims.

In the drawings:

FIG. 1 is a simplified longitudinal half section of the fan component of a turbofan engine;

FIGS. 2 and 3 diagramatically illustrate air flow relative to a blade at the hub and tip sections and to an angled blade leading edge;

FIG. 4 is a side elevation of the improved blade of the present invention, with the hub portion simplified;

FIG. 5 is a tip end view taken on line 5—5 in FIG. 4;

FIG. 6 is a section taken on line 6—6 in FIG. 4;

FIG. 7 is a section, at the hub profile, taken on line 7—7 in FIG. 4; and

FIG. 8 is an elevation of the blade of FIG. 4, in an untwisted condition.

FIG. 1 illustrates the fan section of a turbofan engine wherein the blades of the present invention find particular utility.

The annular engine inlet, indicated at 20, is defined by an outer cowl 22 and a central spinner 24. Air entering the inlet 20 is pressurized by fan blades 26 projecting radially from a rotor 28. The blades are mounted on the rotor 28 by roots, or tangs, 29, which are restrained in appropriate guides formed in the rotor. Downstream of the blades 26 the pressurized air is split by the leading edge of an engine housing, or nacelle 30.

The outer portion of the pressurized air stream passes through a cascade of stator vanes 32 and then is discharged through an annular nozzle 34 defined by the downstream end of the cowl 22 and the nacelle 30. The inner portion of the pressurized air stream is directed through an inlet 36 to the gas turbine engine, or gas generator.

The gas turbine engine may be of conventional design, comprising a compressor for further pressurizing the air and a combustor in which the pressurized air supports combustion of fuel in the generation of a hot gas stream. A portion of the gas energy is extracted by a turbine to drive the gas generator compressor. A second power turbine extracts a further portion through a nozzle to provide additional propulsive thrust.

The power turbine has a forwardly extending output shaft 38 on which the spinner 24 and rotor 28 are mounted and rotated thereby. The output shaft 38 is journalled, in part, on a frame element 40. It will also be seen that the cowl 22 is structurally connected to the nacelle 30 by angularly spaced struts 42.

The described fan section comprises what may be referenced as a low pressure compressor, or fan, the latter term being used in the following description.

In high bypass ratio engines the fan rotor rotates at a rate such that the air velocity relative to the upper portion of the blades is supersonic and forward propagating shock waves are generated along the leading edges of conventional blades. These shock waves generate the "multiple pure tone noise" referred to above. The blade of the present invention minimizes this objectionable noise.

Before describing this improved blade, a brief reference will be made to some basic considerations of blade design, with reference to FIG. 2. Broken line *a* indicates the rotor axis and arrow *b*, the sense of rotation of a blade whose hub profile is shown by a solid outline. Vector V_x represents the rotational velocity of the leading edge of the blade at the hub profile radius. V_u represents the absolute velocity of air at the leading edge. Vector V_r represents the relative velocity between the air and the blade leading edge at that radius. Typical relative Mach number values are $M_r = 0.8$ at the hub, and $M_r = 1.5$ at the tip sections.

At the hub, the chord line *c* of the blade is set at an angle *t* relative to the axis *a* to provide the desired angle of attack of the vector V_r on the leading edge. The vector V_u remains essentially constant across the width of the fan inlet, while the vector V_x increases in a radially outward direction. Thus, the setting angle *t* generally increases in that direction. This is illustrated by the broken outline of the blade tip profile. At this point it will be pointed out that the term "profile" denotes a blade cross section essentially perpendicular to the radius, while "blade element" denotes blade portion comprised between two closely spaced cross sections.

A fundamental principle in attaining the ends of the present invention is illustrated in FIG. 3. The vector V_r is shown impinging on a radially extending leading edge *d*, normal to air flow direction. The angled broken line leading edge *d'* shown in the same figure demonstrates that the velocity component V_t, normal to the inclined blade edge is substantially smaller than the relative flow velocity V_r.

The blades 26 incorporate this principle, as will be evident from the following description of FIGS. 4–8. FIGS. 4–7 show the blade 26 in its actual twisted configuration. FIG. 8 is for illustrative purposes and shows the blade untwisted and the vectors V_r rotated around a radial axis so that the leading edge and the rotated vectors V_r are in a common plane.

From FIG. 8 it can be seen that the leading edge 44 of the blade 26 is swept forwardly from its hub end out to a point 46 located inwardly from the tip end 48 of the blade. From the point 46 the leading edge 44 is swept rearwardly to the tip end 48. The forwardly and rearwardly swept portions of the leading edge follow a curve which has a progressively decreasing slope angle θ with respect to the direction of air flow, as indicated by the vector V_r.

It will be seen that the velocity components V_{t1} and V_{t2} normal to the leading edge 44, at points inwardly and outwardly of sweep reversal point 46, are substantially smaller than the corresponding velocities V_{r1} and V_{r2}.

The slope of the leading edge is such that the component V_t is maintained below the sonic velocity at which objectionable forward propagating noise would be generated. The slope angle θ thus is progressively decreased, as will be apparent from the angles θ_1 and θ_2.

Further, depending on the design requirements of a given blade the slope of the leading edge 44 may be decreased to maintain the normal velocity component V_t below the critical value along a portion or the entire length of the leading edge.

At the design speed of blade 26 the velocity V_r at the hub profile is in excess of the critical or the sonic value and the leading edge is swept from the point outwardly to the tip end of the blade. In some cases the relative air velocity at the hub profile will be below the sonic velocity or the critical velocity, or both. In such case sweeping of the leading edge would not necessarily begin

until the radially outward point on the leading edge where the critical or sonic velocity is attacined.

Blade profiles, although of an airfoil shape, can vary widely according to the energy to be imparted to the air stream and other parameters well known to those skilled in the art. It is preferred that the profiles of the blade 26 be established in accordance with such known design principles consistent with the teachings herein.

In one sense the present invention involves shifting the tip end blade profile rearwardly and then forming the leading edge, as above described. Broken line *e* in FIG. 8 represents the leading edge of a blade with conventional profiles plotted from the trailing edge 50. It will be seen that the present invention involves increasing the chord lengths of the blade profiles, resulting in the addition of a triangular portion in advance of the line *e*.

From FIGS. 5–7 it can be seen that the setting angle *t* increases and the camber decreases from the hub to the tip end of the blade in accordance with the basic requirements of a conventional design.

In conventional blades the centers of gravity of the blade profiles may be stacked on a radius extending through the center of gravity of the hub section, so that there are no centrifugal bending moments induced in the blade. This same principle could theoretically be employed with the present blade by reversely sweeping the trailing edge to maintain the centers of gravity of all blade profiles radially aligned with the center of gravity of the hub section. Such a blade, however, would generally involve unacceptable weight and aerodynamic penalties.

From purely a weight standpoint it would be desirable to use conventional chord lengths from hub to tip. However, to do so would result in unacceptable bending stresses.

To minimize these bending stresses the trailing edge 50 is smoothly curved from the hub to the tip profiles. This means that the centers of gravity of the blade profiles progressively shift in a direction generally opposite to the direction of air flow from the hub section out to the point of sweep reversal. Then from the point of sweep reversal outwardly to the tip section the centers of gravity progressively shift in the same general direction as the direction of air flow. Bearing in mind that the blade is twisted from hub to tip, the centers of gravity are progressively displaced relative to the major and minor axes of inertia, *x* and *y*, respectively, of the hub section. This is illustrated by the profile centers of gravity *cg* shown in FIGS. 5–7.

Preferably, the summation of the bending moments of the centrifugal forces of the blade elements is essentially balanced relative to the major and minor axes of inertia of the hub section. Thus, there is little or no added bending stress at the hub section where the centrifugal tensile stresses are greatest. This end is attained by positioning the centers of gravity of the blade profiles so that their radial projections straddle the major and minor axes of the hub section. Thus, it will be seen that the centers of gravity are progressively displaced, from the hub section to the point of sweep reversal, to an extreme position to one side of both the major and the minor axis of inertia, *x* and *y*. Then, as the centers of gravity are shifted in the opposite direction outwardly of the point of sweep reversal, an extreme position is reached at the tip section where the center of gravity is on the opposite side of the major and minor axis of inertia, *x* and *y*.

In a further improved configuration the centers of gravity of the blade profiles are positioned so that the summation of the centrifugal moments essentially balances the moment of the aerodynamic blade forces at the hub section.

While it is possible to essentially eliminate bending stresses at the hub section, there will be bending stresses intermediate the hub and tip sections. Generally, these bending stresses will be greatest in the trailing edge region at the point of sweep reversal and this will dictate the minimum profile chord length and thickness at that location.

In determining the configuration of the blade 26 the first step is to define the hub and tip section profiles according to conventional aerodynamic design procedures. The point of sweep reversal is then tentatively selected and the slope of the leading edge line is established so as to maintain the component of the air velocity normal thereto below the sonic or critical value over the entire span of the blade. This can be done in many ways, since the leading edge line in principle must only satisfy the condition that its tangent at every point lie on a cone whose axis coincides with the relative velocity direction at that point, whose vertex is at that point and whose aperture is such that the projection of the relative velocity perpendicular to the cone surface is smaller than the sonic or critical velocity. For the initial attempt the leading edge line is selected so that the straddling condition of the centers of gravity of the tip and the sweep reversal profiles relative to the major and the minor axis of inertia of the hub section is qualitatively satisfied. A trailing edge line is then established to define intermediate profiles with minimum chord length and adequate thickness. It is then possible to determine the centrifugal force of the blade and its bending moments relative to the major and minor axes of inertia of the hub section. This process is then iterated, i.e. the location of the point of sweep reversal, the position of the centers of gravity of the profiles and the trailing edge line are modified until those moments are minimized or essentially balance the aerodynamic moment at the hub section.

If, after optimization of the point of sweep reversal, as above described, there remains a substantial imbalance of the centrifugal moment relative to any other blade section, then the iterative process of positioning the centers of gravity of the profiles must be pursued until proper straddling of their radial projections relative to the minor axis of inertia, *x*, of that section is achieved, while preserving the balanced conditions at the hub section. This in general will require additional shifts of the centers of gravity of profiles located below as well as above the intermediate section involved.

During those final iterations, the profile chord lengths and thicknesses may be adjusted at particular locations to achieve a complete design optimization.

The described blade and the method of determining its configuration essentially eliminates standing shock waves on the leading edge and the objectionable noise generated thereby. Further, there is a minimum increase in weight to attain this primary end, while at the same time stresses are maintained at safe levels when operating at the peripheral speeds of advanced high bypass ratio fans.

It is to be emphasized that the foregoing description of the blade 26 illustrates only a particular design example and the basic method and steps to be followed in the design of a blade which essentially eliminates stand-

ing shock waves along its leading edge with minimum additional weight and stresses.

Variations from the embodiments and the method herein described will occur to those skilled in the art within the spirit and scope of the present invention which are to be limited solely by the following claims.

Having thus described the invention what is claimed as novel and desired to be secured by Letters Patent of the United States is:

1. In an axial flow compressor:

a rotor having a circumferential row of radially projecting blades, each blade having means at its hub end for attaching it to the rotor,

each blade being in the form of an airfoil having a progressively decreasing camber and progressively increasing angle of twist from its hub end to its tip end,

said rotor having a rate of rotation at which at least an outer portion of the blade has a velocity, relative to the air flow therepast, which is at a sonic value or greater, said relative velocity being sufficient to generate a shock wave on the leading blade edge disposed normal to the direction of relative airflow,

characterized in that

the profiles of each blade at the inner, or hub end, and at the outer or tip end of said outer portion are of conventional length,

the leading edge of said outer portion is swept forwardly from its hub end on a curve having a progressively decreasing slope relative to the direction of air flow, outwardly to a point of sweep reversal spaced inwardly from the tip end of the blade and, from said point of reversal, the leading edge is swept rearwardly to the tip end of the blade on a curve also having a progressively decreasing slope relative to the direction of air flow therepast, said slope at all points along the forwardly and rearwardly swepted portions of the leading edge being such that the flow velocity component normal thereto remains below a sonic value, and

the trailing edge of said outer portion of the blade is smoothly curved in a rearward direction from its hub end to its tip end, whereby the chord length of the blade profiles progressively become greater than conventional lengths outwardly to said point of sweep reversal and then the profile chord lengths progressively decrease to the conventional length profile at the tip end of the blade, and

further characterized in that

the point of curvature reversal is so disposed that the centers of gravity of the blades profiles progressively shift in one direction relative to the major and minor axis of inertia of the attachment means from the hub end of said outer portion to said point of sweep reversal and then shift in the opposite direction, relative to said axis of inertia, from the point of sweep reversal to the tip end, a distance such that a summation of the centrifugal bending moments is essentially balanced relative to the attachment means due to the straddling of the axis of inertia by the centers of gravity of the blade profiles.

* * * * *

7

United States Patent [19]

Weingold et al.

[11] Patent Number: 4,726,737

[45] Date of Patent: Feb. 23, 1988

[54] REDUCED LOSS SWEPT SUPERSONIC FAN BLADE

[75] Inventors: Harris D. Weingold, West Hartford; Walter B. Harvey, South Glastonbury, both of Conn.

[73] Assignee: United Technologies Corporation, Hartford, Conn.

[21] Appl. No.: 924,007

[22] Filed: Oct. 28, 1986

[51] Int. Cl.⁴ B63H 1/26

[52] U.S. Cl. .... 416/223 A; 416/DIG. 2

[58] Field of Search .... 416/223 R, 228, 223 A, 416/DIG. 2

[56] References Cited

U.S. PATENT DOCUMENTS

2,043,736 6/1936 Charavay 416/DIG. 2
2,258,795 10/1941 New 416/DIG. 2
4,047,841 9/1977 Laurin .... 416/223 R
4,431,376 2/1984 Lubenstein et al. 416/DIG. 2
4,569,631 2/1986 Gray 416/DIG. 2

FOREIGN PATENT DOCUMENTS

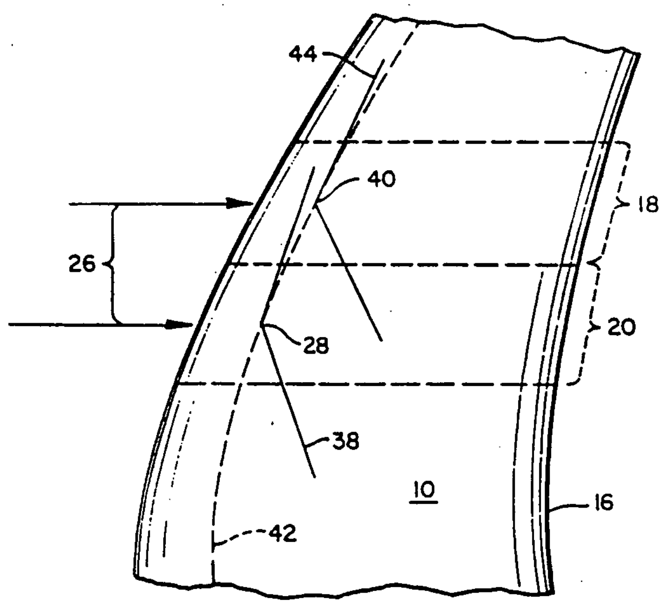
266475 7/1976 U.S.S.R. 416/DIG. 2

Primary Examiner—Robert E. Garrett
Assistant Examiner—John T. Kwon
Attorney, Agent, or Firm—Troxell K. Snyder

[57] ABSTRACT

A swept leading edge blade (10) rotates in a stream of air (26) and experiences supersonic relative air velocities over at least a portion of the blade surface. The maximum camber points (28, 40) of adjacent blade segments (20, 18) are located to prevent intensification of the compression wave generated by each segment (20, 18).

2 Claims, 4 Drawing Figures



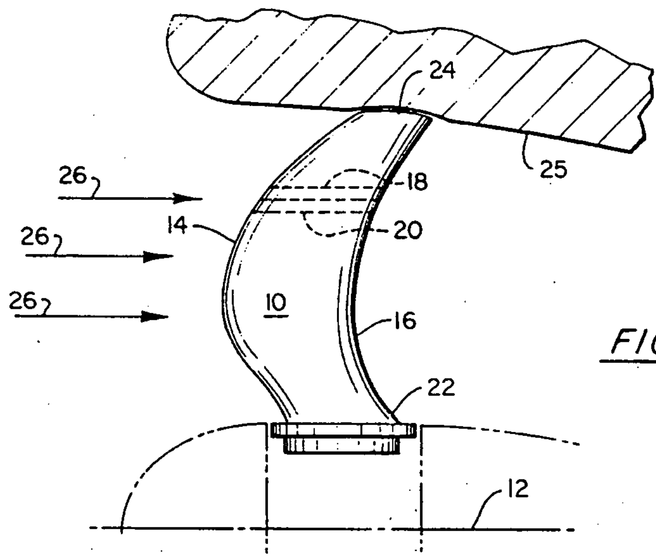


FIG. 1

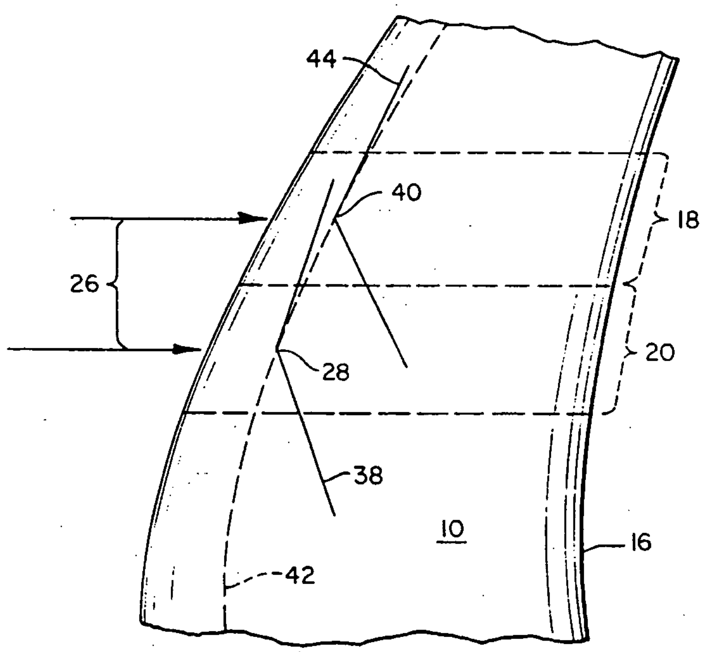


FIG. 2a

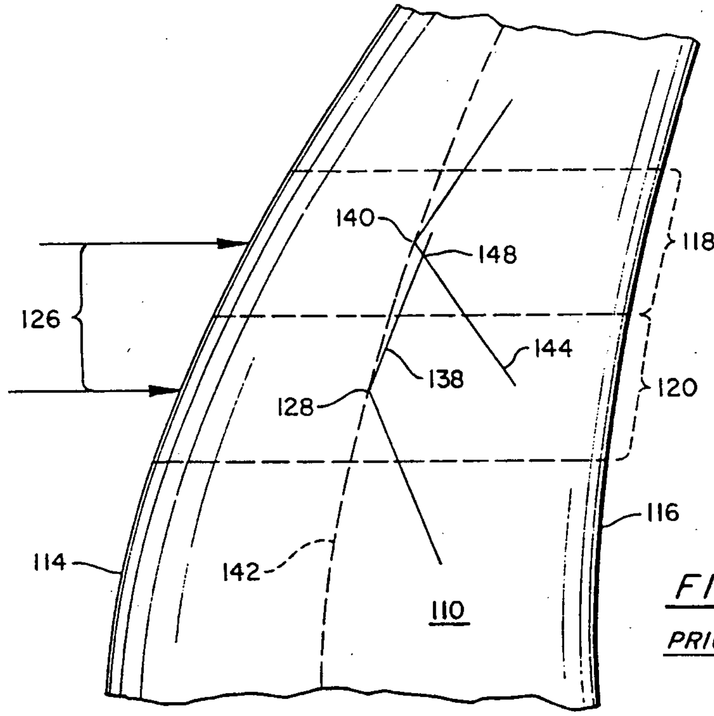


FIG. 2b
PRIOR ART

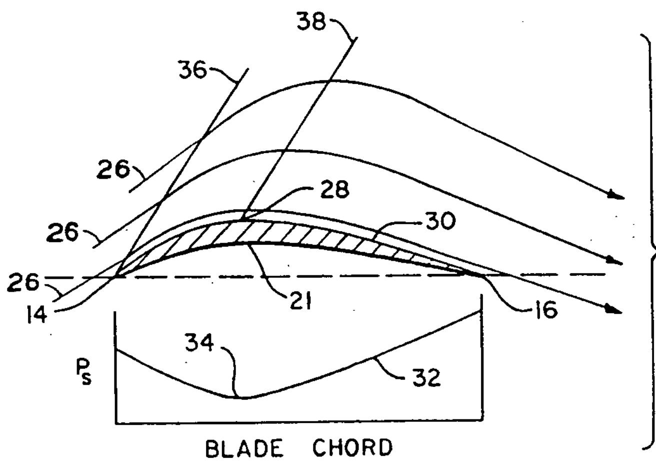


FIG. 3

REDUCED LOSS SWEPT SUPERSONIC FAN BLADE

FIELD OF THE INVENTION

The present invention relates to a fan or compressor blade having supersonic airflow over at least a portion of the blade span.

BACKGROUND

The occurrence of supersonic flow over a portion of a rotating compressor or fan blade is a phenomenon which is familiar to those skilled in the art of aircraft propulsion system design. Relative flow velocity is a function of the engine nacelle or inlet through flow velocity, the angular velocity of the rotating blade, and the radial distance between the blade region under consideration and the axis of rotation.

Prior art blade designers have recognized the flow disruption which results from such relative supersonic flow, and, in particular with respect to noise generation, have specified blade leading and trailing edge shapes responsive to the supersonic flow conditions which attempt to minimize the occurrence of sonic shock waves in the vicinity thereof. One such blade design is shown in U.S. Pat. No. 3,989,406 issued Nov. 2, 1976 to Bliss. The Bliss reference shows a rotor blade wherein the leading edge of that portion of the blade subject to supersonic relative airflow velocities is swept axially forward or rearward behind a Mach cone defined at each point along the blade leading edge. This critical skewing or sweeping of the blade leading edge results in the component of the relative airflow velocity normal to the leading edge of the blade having a Mach number less than 1 and therefore falling in the subsonic flow regime. The Bliss design is intended to prevent leading edge shock waves from forming thereby reducing shock related noise generation by the rotating blades.

Bliss also discusses the problem of the formation of pressure waves adjacent the shroud wall surrounding the rotating blades. These shroud wall pressure waves interact with the airflow over the blade tips and can result in a second shock wave in the vicinity of the rotating blade tips. Bliss further shows a method and means for reducing such shroud-tip interaction by contouring the shroud wall along the natural streamline deflection over the suction surface of the blade tip.

A related blade design is disclosed in U.S. Pat. No. 4,012,172 issued Mar. 15, 1977 to Schwaar et al. The Schwaar reference shows a noise reducing fan blade design wherein the leading edge of the blade is swept forwardly from the blade hub to a point at the blade midspan and then swept rearwardly to the blade tip. The leading edge sweep is, for that portion of the blade subject to supersonic relative airflow velocities, shaped to fall within the Mach cone of the upstream adjacent leading edge points, thus achieving the subsonic normal velocity component indicated in the Bliss reference. Schwaar also describes a method for configuring such blades to minimize internal blade bending and attachment stresses by balancing the centers of gravity of the successive blade transverse segments about the attachment radius such that radial forces induced by the movement of the blade about the rotation axis are substantially balanced.

Finally, U.S. Pat. No. 4,358,246 issued Nov. 7, 1982 to Hanson et al also shows the contouring of a supersonic prop fan blade so that both the leading and trailing

edges of the blade are swept behind their corresponding Mach surfaces. The pressure spike induced in prior art blades by the trailing wave is thus minimized along with the noise generation resulting therefrom.

The prior art blade designs thus discussed have been directed toward minimizing the noise generation which can occur in the case of improperly designed supersonic fan blades. Such noise reducing designs as are present in the prior art, while effective in reducing leading and trailing edge shock waves, do not address the occurrence of a strong compression wave front at the suction surface of a supersonic blade intermediate the leading and trailing edges. The compression wave front is the result of the recompression of the air stream flowing over the suction surface of the fan blade as the local airflow ceases its relative acceleration with respect to the blade surface and begins to decelerate adjacent that portion of the blade which lies downstream of the point of maximum blade camber.

As airflow relative velocity decreases, the static air pressure rises from the minimum value which occurs coincident with the maximum relative airflow velocity. This increase in pressure forms a pressure wave which, reinforced by the pressure waves of adjacent points along the blade suction surface, contributes to the creation of a rapid static pressure jump wherein the pressure gradient at the blade surface rises too rapidly to permit smooth flow in the boundary layer downstream thereof. This strong compression wave front induced by the reinforced compression waves at the surface of the blade detaches and disrupts the downstream blade surface boundary layer, resulting in flow recirculation and other irreversible losses which diminish overall blade performance by both increasing blade drag and decreasing the amount of airflow turning or static pressure rise achieved.

Prior art supersonic blade designs have attempted to delay such separation and its deleterious performance effects by shaping the transverse cross section of individual blade segments such that the point of maximum blade camber is located toward the trailing edge of the blade, with the exact location for each blade section based on prior state of the art unswept blade cascade test data correlations. The separation of the boundary layer which occurs due to the recompression of the flowing air downstream of the maximum blade camber point is thus able to act on only a small portion of the blade suction surface.

The prior art technique, while effective to a degree, does not eliminate the disruption of the boundary layer and the resulting inefficiencies and irreversibilities, but only reduces the total negative effect thereof. What is required is a blade design which weakens or eliminates the compression wave front to such a degree that separation of the airflow boundary layer at the suction surface of the rotor blade does not occur.

DISCLOSURE OF THE INVENTION

It is therefore an object of the present invention to provide a blade for a rotating fan wherein the relative velocity between the air and at least a portion of the blade is supersonic.

It is further an object of the present invention to provide a blade configured to avoid separation of the suction surface boundary layer during recompression of the air stream.

It is still further an object of the present invention to eliminate the compression wave front resulting from interaction of individual compression waves generated in adjacent blade segments.

According to the present invention, a rotating blade for inducing an axial static pressure rise resulting in a propulsive force is configured to achieve both the low noise generation of prior art designs and the high efficiency operation demanded by modern aircraft operators. High efficiency is achieved through configuring individual airfoil segments of the blade so as to minimize or eliminate the strong compression wave front occurring at the surface of prior art blades and which results in separation of the downstream air boundary layer from the suction surface of the blade. Boundary layer separation, particularly in lifting airfoil or propulsion blade applications, results in irreversible (and hence inefficient) gas flow and increased drag on the airfoil.

The blade according to the present invention avoids creating a strong compression wave front by locating the maximum camber points of individual blade airfoil segments such that the compression wave generated by any one segment of the rotating blade is not reinforced (and therefore not strengthened) by the compression wave generated by any other blade segment. By avoiding the intersection and reinforcement of the compression waves generated at any point along the blade span, the blade according to the present invention reduces or eliminates the boundary layer separation common in prior art blades operating at supersonic relative speeds.

More specifically, the location of maximum camber of the individual blade airfoil segments of the blade according to the present invention is located between the blade leading and trailing edges such that the compression wave occurring immediately downstream of the point of lowest static pressure of each blade segment is disposed within a Mach cone defined at the equivalent point of the upstream adjacent airfoil segment. The local effect of the recompressing air flow of each segment is thus swept back along the blade surface at a rate which prevents the compression wave generated at any one point on the blade from intersecting and thus reinforcing the pressure wave generated at any other point in the blade.

For those blades configured with swept leading edges for avoiding the generation of a leading edge shock wave, the line of maximum camber defined by the maximum camber points of the individual blade airfoil segments is similarly swept at least over those portions of the blade span which experience supersonic relative airflow velocities.

As rotating blades for gas turbine engine fans or prop fan applications typically taper in thickness and vary in chordal dimension with radially outward displacement, the blades according to the present invention have a varying proportional chordal displacement of the maximum camber point of each airfoil section over the span of the blade in order to achieve the swept compression wave described above. This represents a significant departure from prior art blade designs wherein a "family" of airfoil patterns are typically used to define the individual blade segments and which, although changing in relative thickness, chordal length, and maximum relative camber over the span of the blade, still retain a characteristic and nearly constant proportional displacement between the maximum camber point and the blade leading edge relative to the segment chord length. Such prior art blades thus create compression waves at each segment which intersect with compression waves generated by downstream segments resulting in an intensified compression wave front which in turn disrupts the airflow over the suction side of the blade surface and diminishes blade aerodynamic efficiency.

Both these and other objects and advantages will become apparent to those skilled in the art upon review of the following description and the appended claims and drawing figures.

BRIEF DESCRIPTION OF THE DRAWINGS

FIG. 1 shows a side view of a blade according to the present invention in the plane of the rotation axis.

FIG. 2a shows a more detailed view of a blade according to the present invention indicating the location of the compression waves generated by individual blade segments.

FIG. 2b shows a prior art blade wherein the compression waves generated by individual blade segments intersect and reinforce.

FIG. 3 shows a transverse cross section of a blade according to the present invention indicating both local air stream lines and the airfoil suction surface static pressure distribution.

DETAILED DESCRIPTION OF THE PREFERRED EMBODIMENT

FIG. 1 shows a side elevation of a fan blade rotating about a central axis 12. The blade 10 includes a swept leading edge 14, a trailing edge 16, and is comprised of a plurality of individual blade airfoil segments 18, 20 disposed between the radially inward root end 22 and the radially outward tip end 24. The fan is enclosed in an annular shroud or duct 25.

It should be noted that the blade segments 18, 20 represent arbitrary divisions of the blade 10 made for computational convenience when the blade performance and local airflow is analyzed by a finite difference numerical method or the like. The thickness of an individual segment or "slice" is thus dependent on the computational power of the analytical method employed, with the more powerful methods being typically able to accommodate a greater number of thinner segments and thus achieving a higher degree of overall accuracy as a result. An exact mathematical solution of the blade airflow would define the relationship between differential blade segments of infinitesimal thickness, however this type of solution has not yet been achieved for any but the most simple of airfoil arrangements. Current numerical methods of analysis are able to compute the local airflow conditions in 20 to 50 segments distributed radially over the blade span between the root and tip. Although discussed in terms of individual adjacent blade segments having a finite thickness, it is therefore to be understood that the term "segment" is intended to include the minimum computational "slice" of the overall blade consistent with the method of analysis employed.

FIG. 1 also shows an axially flowing stream of air 26 encountering the swept leading edge of the blade 14. The degree of sweep, both axial and circumferential, of the leading edge 14 is related to the relative velocity between the leading edge 14 and the air stream 26 and is such that the relative velocity component normal to the leading edge 14 is less than the sonic velocity under airflow conditions local to that point. Such designs are well known in the prior art and eliminate the formation of sonic shock waves at the leading edge of blades such

as those shown in FIG. 1 which have at least a portion thereof operating in a supersonic flow regime relative to the corresponding air stream.

As will be appreciated by reviewing FIG. 3 which shows a cross section 21 of airfoil segment 20. of the blade according to the present invention, the section 21 has a generally curved shape extending between the leading edge 14 and the trailing edge 16. The cross section 21 defines a curved or cambered volume having a point of maximum camber 28 located intermediate the leading and trailing edges 14, 16 on the suction or convex surface 30 of the airfoil segment 20.

As is well known in fluid theory and shown by curve 32 in FIG. 3, the static pressure of the airflow 26 over the blade segment 20 decreases as the airflow 26 is turned by contact with the segment 20 until the flow passes the point of maximum camber 28 at which time the airflow velocity begins to decrease and the gas stream recompresses as shown by curve 32. The point of minimum static pressure 34 also represents the local compression wave front initiated at the surface 30 of the airfoil segment 20.

As is also well known in the art of supersonic and compressible fluid flow, the effect of any perturbation in a supersonic flow regime is concentrated along a "Mach line" (or Mach cone in three-dimensional flow) which is a function of the Mach number of the flow stream according to the relationship

$$\alpha = \text{arc sine } 1/M$$

wherein

M equals the Mach number of the flow stream, and

α equals the included half angle of the Mach cone.

Thus, for the two dimensional representation of FIG. 3, the perturbation effects of the leading edge 14 and maximum camber point 28 of the segment 20 are transmitted into the gas flow stream 26 along respective Mach lines 36, 38. Due to the nature of supersonic flow, not only is the effect of a flow perturbation not evident in the airstream 26 upstream of the respective Mach line 36, 38, but the disturbance caused by the surface perturbation 14, 28 is concentrated along the respective Mach line 36, 38. Thus, the increase in pressure or compression wave generated at the maximum camber point 28 for the airfoil segment 20 propagates to adjacent segments and into the airstream 26 principally along the Mach line or cone 38.

The significance of the foregoing discussion is best appreciated by reviewing FIGS. 2*a* and 2*b* in concert. FIG. 2*a* shows a detailed view of the segments 18, 20 of the blade 10 of FIG. 1. The maximum camber point 28 of the segment 20 and a similar maximum camber point 40 of segment 18, along with the maximum camber point of the other segments comprising the blade 10 define a maximum camber line 42 as shown in FIG. 2*a*. As discussed hereinabove, the maximum camber point 28 is related to the minimum static pressure point 34 occurring on the suction surface of the segment 20 as well as the initiation of the recompression of the air flow 26 flowing over that segment 20. The effects of this recompression propagate spanwise along the blade 10 along the Mach lines 38, 44 for the respective maximum camber points 28, 40.

The individual segments 18, 20 of the blade 10 according to the present invention are cambered responsive to the relative velocity of the airflow 26 and the Mach lines 38, 44 created at the maximum camber points 28, 40 such that the compression wave initiated at

the maximum camber point 40 of segment 18 lies behind the Mach line 38 and hence the compression wave generated at point 28 on the upstream blade segment 20. The compression waves generated by each blade segment 18, 20 are thus swept behind the Mach lines of the adjacent upstream blade segment compression waves and do not intersect or reinforce each other.

This advantage is best appreciated by reviewing FIG. 2*b* wherein a prior art blade 110 having adjacent blade segments 118 and 120 experiences relative supersonic flow with respect to an airstream 126. The maximum camber points 128 and 140 of the respective segments 120 and 118 define a maximum camber line 142 as shown and are not arranged such that the downstream blade segment maximum camber point 140 is located behind the Mach line 138 originating at the maximum camber point 128. The effects of the compression waves generated at the maximum camber points 128 and 140 thus propagate along the Mach lines 138 and 144 as shown, intersecting at point 148 and forming a reinforced wave front behind the maximum camber line 142.

The intersection of the propagating compression waves from adjacent blade segments 118, 120 provides a reinforced and intense compression wave front immediately downstream of the maximum camber line 142 and results in a rapid jump in surface static pressure along the suction surface of the blade 110. Such a rapid pressure jump or shock wave adjacent the blade surface can disrupt and detach the downstream boundary layer and diminish the aerodynamic effectiveness of the local blade surface. The irreversibilities caused by the occurrence of such a shock wave and boundary layer detachment transfer energy from the rotating blade into the airstream 26 as nonuseful heat energy rather than the desirable airflow turning or static pressure rise achieved by the blade according to the present invention over the entire blade surface.

The swept maximum camber line 42 of the blade 10 according to the present invention would appear in most applications to be somewhat uniformly spaced from the leading edge 14 as it follows a similar design parameter. Since typical fan blades are formed of airfoil sections 18, 20 which vary with respect to chordal dimensions with increasing radius, the blade 10 according to the present invention is constructed of airfoil segments having a non-uniform proportional displacement between the maximum camber point and the leading edge with respect to the segment chordal dimension. This is a significant departure from prior art blade designs which are typically constructed of airfoil segments based upon a plurality of similar airfoil shapes or "family" wherein the location of the maximum camber point of each segment is a substantially constant proportion of the segment chordal dimension, thereby insuring that the maximum camber line 142 will not be uniformly spaced from the sonically swept leading edge 114.

It will be appreciated that the actual design and position of the maximum camber points 28, 40 and maximum camber line 42 of the blade 10 according to the present invention are located responsive to the local airflow over the particular segment under consideration and, although this may typically result in a substantially uniform spacing between the maximum camber line 42 and the leading edge 14, it does not in any way require such a relationship which is simply the likely result of the location of the maximum camber points 28, 40 so as

to result in the non-intersection of the compression waves generated at each and every segment along the blade span.

This local airflow must be determined through the use of a three dimensional transonic flow analysis procedure rather than two dimensional modeling or cascade test data which does not accurately account for spanwise or radial flow induced by adjacent blade segments. Such three dimensional analysis techniques are particularly important near the blade tip where the blade camber distribution must be determined simultaneously with the shroud contour to avoid creating a coalescence of pressure waves leading either to a secondary shock, as described in the Bliss reference, U.S. Pat. No. 3,989,406 discussed above, or to a premature detachment of the boundary layer from the blade or shroud surfaces.

It will further be appreciated that the relationships between the adjacent segments as disclosed and described hereinabove are intended to only cover that portion of the rotating blade wherein the relative velocity between the moving blade and moving air stream is greater than the sonic velocity under local air temperature and flow conditions.

Although disclosed in terms of a rotating fan blade swept primarily rearwardly from the hub and to the tip end, the blade configuration according to the present invention is equally beneficially applicable to blades of forward sweep, reversing sweep, as well as ducted and unducted fan propulsion systems, inducers of centrifugal impellers, and prop fan blades. The foregoing discussion is therefore to be taken in an illustrative sense and shall not be construed as limiting the scope of the invention as defined solely by the claims recited hereinbelow.

I claim:

1. A fan blade rotating about a central axis at an angular velocity sufficient to result in a supersonic relative velocity between a portion of the blade and an axially flowing stream of air, including a leading edge, a trailing edge, a plurality of blade airfoil segments forming a suction surface extending between the leading edge and the trailing edge, each segment defining a point of maximum camber and a corresponding minimum static pressure point on the suction surface, the segments collectively defining a blade having a span extending from a radially inward root end to a radially outward tip end, and the points of maximum camber collectively defining a maximum camber line along the suction surface of the blade,

characterized in that

the line of maximum camber in that portion of the blade experiencing supersonic relative velocity is swept with respect to the airflow such that any one minimum static pressure point on the blade suction surface falls within a Mach cone defined at any other minimum static pressure point lying both on the blade suction surface and upstream of the one minimum static pressure point.

2. In a rotating blade for interacting with a stream of air flowing substantially parallel to the blade axis of rotation, the relative velocity between the airstream and at least a portion of the blade being supersonic, the supersonic portion of the blade further including a plurality of cross sectional airfoil span segments, each segment having a leading edge and a trailing edge, and interacting with the airflow stream to locally create a corresponding minimum static air pressure point on the suction surface of each airfoil span segment intermediate the leading and trailing edges thereof, wherein the improvement comprises:

each airfoil span segment being located with respect to the next downstream adjacent segment such that the leading edge point of the downstream segment falls within a first Mach cone defined by the leading edge of the adjacent upstream segment, and

the minimum convex surface static air pressure point of the downstream adjacent segment falls within a second Mach cone defined at the minimum suction surface static air pressure point of the upstream adjacent segment.

* * * * *

8

United States Patent [19]

Bliss

[11] **3,989,406**

[45] **Nov. 2, 1976**

[54] **METHOD OF AND APPARATUS FOR PREVENTING LEADING EDGE SHOCKS AND SHOCK-RELATED NOISE IN TRANSONIC AND SUPERSONIC ROTOR BLADES AND THE LIKE**

[75] Inventor: **Donald B. Bliss**, Belmont, Mass.

[73] Assignee: **Bolt Beranek and Newman, Inc.**, Cambridge, Mass.

[22] Filed: **Nov. 26, 1974**

[21] Appl. No.: **527,499**

[52] **U.S. Cl.** **415/1**; 415/181; 416/228

[51] **Int. Cl.**[2] **F01D 5/12**

[58] **Field of Search** 415/181; 416/228, 223

[56] **References Cited**

UNITED STATES PATENTS

2,582,107	1/1952	Dakin	416/237
2,663,493	12/1953	Keast	416/222
3,721,507	3/1973	Monteleone	416/228 X

FOREIGN PATENTS OR APPLICATIONS

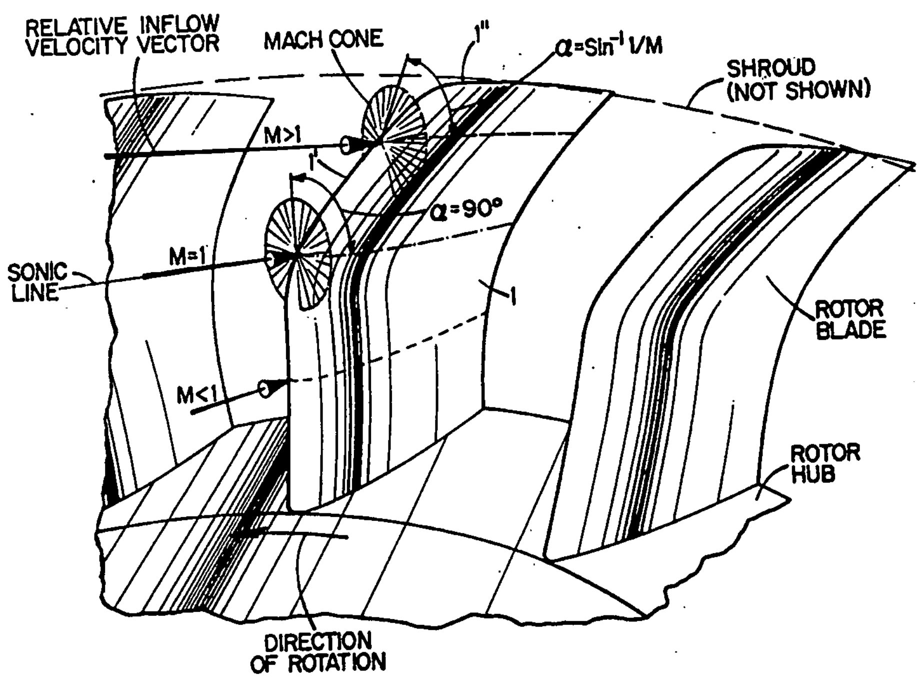
719,523	10/1965	Canada	415/181
719,457	12/1954	United Kingdom	415/181

Primary Examiner—Arnold Rosenthal
Attorney, Agent, or Firm—Rines and Rines

[57] **ABSTRACT**

This disclosure is concerned with novel rotor blade and similar foil designs that, by critical skewing of intermediate blade regions where the airflow is supersonic, prevents leading edge shocks and shock-related noise.

11 Claims, 6 Drawing Figures



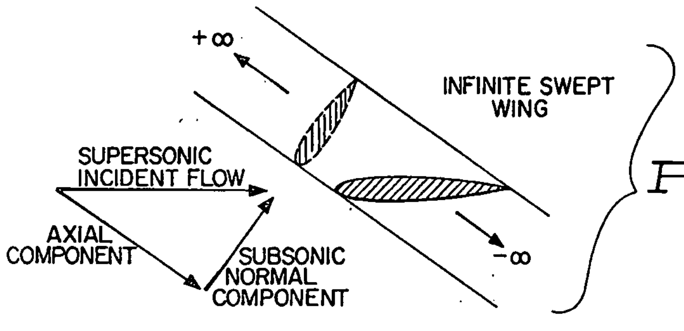
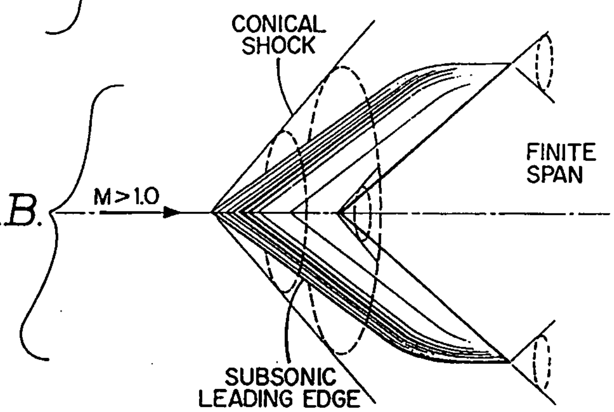


Fig. 1A.

Fig. 1B.



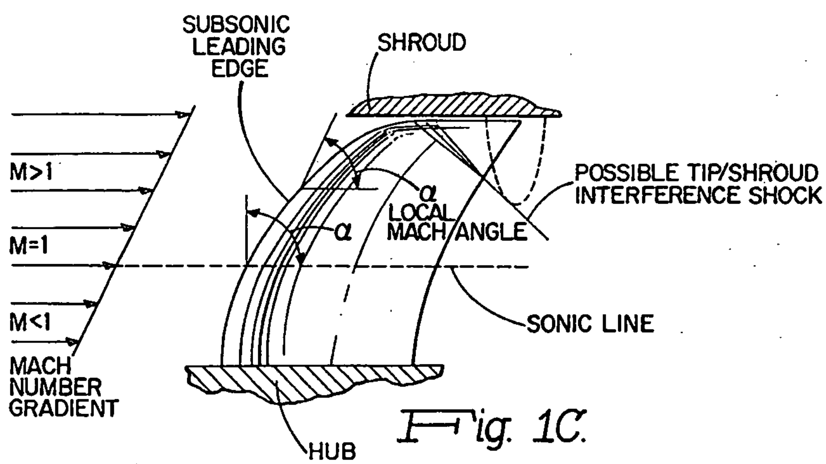
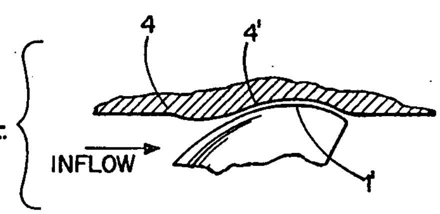


Fig. 1C.

Fig. 4.



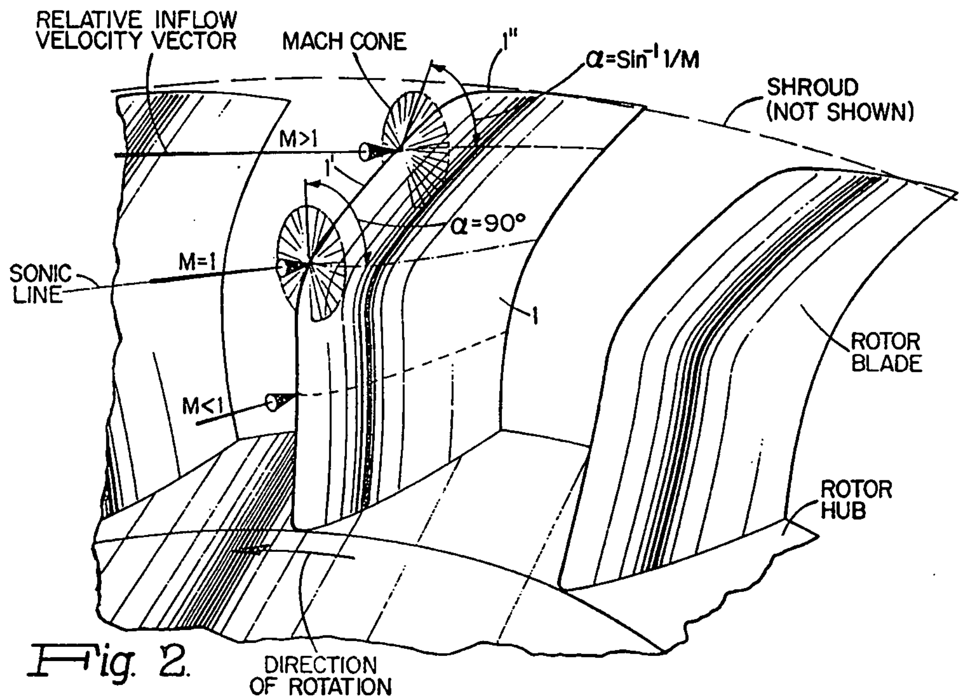

RELATIVE INFLOW VELOCITY VECTOR
MACH CONE
1"
$\alpha = \sin^{-1} 1/M$
SHROUD (NOT SHOWN)
M>1
1'
$\alpha = 90°$
SONIC LINE
M=1
1
ROTOR BLADE
M<1
ROTOR HUB
DIRECTION OF ROTATION

Fig. 2.

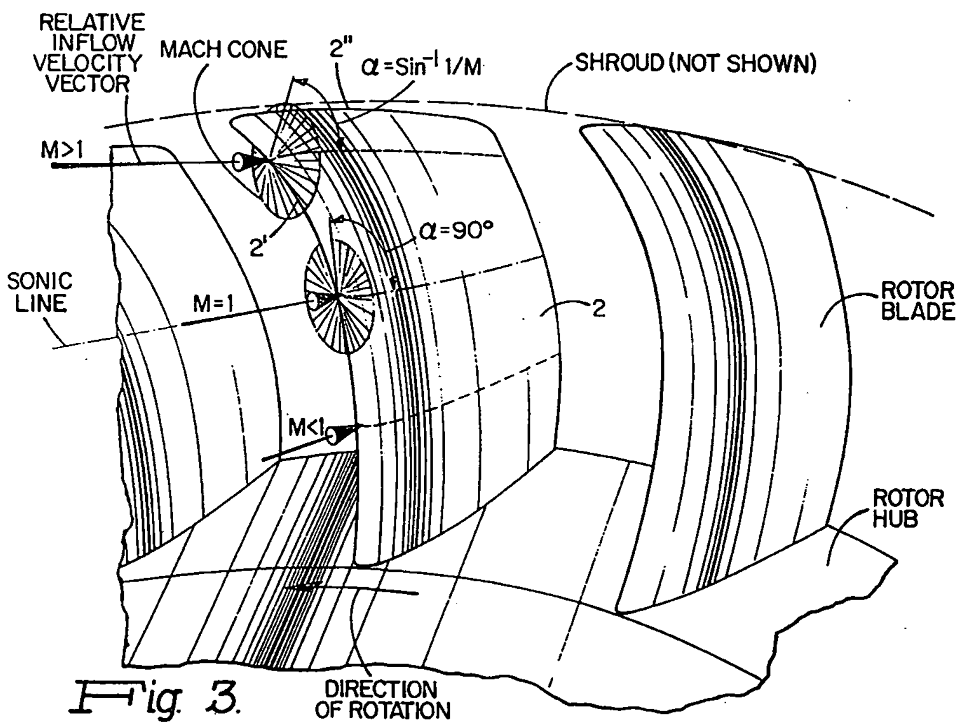

RELATIVE INFLOW VELOCITY VECTOR
MACH CONE
2"
$\alpha = \sin^{-1} 1/M$
SHROUD (NOT SHOWN)
M>1
2'
$\alpha = 90°$
SONIC LINE
M=1
2
ROTOR BLADE
M<1
ROTOR HUB
DIRECTION OF ROTATION

Fig. 3.

METHOD OF AND APPARATUS FOR PREVENTING LEADING EDGE SHOCKS AND SHOCK-RELATED NOISE IN TRANSONIC AND SUPERSONIC ROTOR BLADES AND THE LIKE

The present invention relates to apparatus and methods of preventing or materially reducing leading edge shocks and shock-related noise in transonic and supersonic rotor blades and similar foils (hereinafter generally referred to as "blade" and sometimes as "wing" or "foil"), being more particularly, though not exclusively, concerned with turbofans and similar jet engine structures.

Many modern aircraft use high-bypass ratio turbofan engines for propulsion, in which the first stage consists of a large multi-bladed fan rotor followed by a set of fixed stator blades and enclosed within a shroud as described, for example, in Jane's "All the World's Aircraft", 1972–73, McGraw-Hill, New York. This fan stage contributes directly to propelling the vehicle because a large portion of the thru-flow bypasses the rest of the engine and is used directly to produce thrust. For reasons of efficiency and maximum output, it is desirable to operate the fan at rotational speeds which make the velocity of the outer portion of the fan blades supersonic relative to the flow; but, unfortunately, this gives rise to a pattern of shockwaves and expansion waves produced by the flow over the blades. Waves produced at the leading edge and on part or all of the suction side of each blade can travel upstream and thus radiate from the shroud inlet to produce shock-related noise as an inherent and natural consequence of the steady-state operation of the rotor. Such noise generation differs from the usual subsonic noise sources which arise primarily from unsteady flow phenomena and which produce fluctuating loads on the blades. Of course, the production of shockrelated noise need not be specifically related to the high-bypass ratio turbofan configuration, since any axial flow turbojet engine or the like having rotor blades which operate at supersonic velocity, relative to the flow over part or all of their blade span, will generate this form of shock-related noise.

The noise caused by shockwaves propagating upstream from the rotor blades is often called combination tone or multiple pure tone (MPT) noise. This form of noise is usually manifested by a series of tones at the blade passage frequency and the shaft rotation frequency and at their harmonics. The tones at the shaft frequency and harmonics arise from blade-to-blade differences due to manufacturing tolerances. Since perturbations owing to blade-to-blade dissimilarities are considerably exaggerated due to a natural instability of the propagating shock wave-train, each fan has its own characteristic MPT noise signature.

MPT noise is not easily reduced by conventional sound treatment procedures; e.g. absorptive lining on the shroud inlet duct as described, for example, in U.S. Pat. Nos. 3,113,634 and 2,759,554-5 and 6. While the strength of the shock field does decay to some extent due to nonlinear attenuation as the waves propagate up the duct, simplified analytical models for this process have shown that the strength of the waves far upstream is quite insensitive to small changes in the initial strength. This suggests that effective reduction of this noise source can be achieved only if the upstream shocks are either eliminated or if their strength is substantially reduced at their source.

The conventional approach to reducing upstream shock strength is based on an essentially two-dimensional idea, as described, for example, by A. Ferri, 1964, "The Supersonic Compressor II — Aerodynamic Properties of Supersonic Compressors", Aerodynamics of Turbines and Compressors, W. R. Hawthorne (ed.), Princeton University Press, pp. 381–397; and by A. W. Goldstein et al, 1973, "Acoustic Properties of a Supersonic Fan", NASA TN d-7096. At each blade cross-section along the supersonic portion of the span, the leading edge and part of suction surface of the blade section are specifically shaped to prevent the formation of waves which could propagate upstream. Typically, there is a sharp leading edge followed by a flat portion of the suction surface designed to be tangent to the relative inflow. In reality, however, there will still be an upstream wave system due to the inevitable non-zero thickness of the leading edges and the presence of a viscous boundary layer; but despite these effects, it is possible to achieve a net reduction in the strength of upstream waves.

While this approach is valid in principle, it has serious shortcomings in practice. Since precise alignment of the blade with the relative inflow is necessary, proper operation occurs only at optimum design conditions. When the fan is called upon to operate under non-optimum design conditions, much of the noise reduction advantage is lost. Such non-optimum design operation is a common occurrence; it is particularly necessary during take-off and landing when, unfortunately, noise reduction is most needed. Furthermore, even for optimum -design operation, it is difficult to know the precise inflow angle over the entire supersonic portion of the blade. The flow phenomena through a rotor are naturally three-dimensional and these three-dimensional effects are especially difficult to predict when the blades have a transonic flow region; i.e. a region where the transition from subsonic to supersonic relative inflow occurs. The presence of a subsonic region near the hub allows pressure signals from downstream of the rotor to travel upstream and distort the supersonic portion of the inflow velocity relative to the blade, such distortion being most pronounced in the transonic and lower supersonic region and being very difficult to predict accurately. Besides this natural inflow distortion effect, additional inflow distortions may also occur at the shroud inlet, caused, for example, by the upwash field of the wing and/or an angle of attack of the shroud under certain flight conditions. Since such distortion may not be axially symmetric, there is no way the blades can be designed to compensate therefor. Finally, because of the critical alignment, very accurate manufacturing techniques are required to assure proper operation of this blade design concept.

From these considerations, it is apparent that a substantial reduction in the stength of upstream shockwaves, or their complete elimination, is required, and without susceptibility to the problems cited above which arise largely as a consequency of the basically three-dimensional character of the flow field in which the rotor operates.

It is accordingly an object of the invention to provide a new and improved method of and apparatus for preventing entirely, or reducing the extent of, leading edge shock-waves and shock-related noise in transonic and supersonic rotor blades and the like that are not subject to the before-described limitations and disadvantages.

A further object is to provide a novel blade or similar apparatus of more general utility, as well.

Other and further objects will be explained hereinafter and are more particularly delineated in the appended claims.

In summary, from one of its broadest aspects, the invention contemplates a blade configuration in which the blade is shaped over the region thereof where the airflow relative to the blade is supersonic, with its leading edge swept, with respect to the direction of airflow at an acute angle less than $\sin^{-1} 1/M$, where M is the Mach number of the supersonic flow. Preferred details are hereinafter presented.

The invention will now be described with reference to the accompanying drawing,

FIGS. **1A**, **1B** and **1C** of which are illustrative flow diagrams illustrating, respectively, the shock-free operation with a swept wing of infinite span when the normal component of airflow is subsonic, the generation of conical shock waves with a finite span swept wing with subsonic leading edges, and a shock-free leading edge of a finite span blade, appropriately provided with a Mach number skew gradient;

FIGS. 2 and 3 are partial isometric views of blades constructed in accordance with preferred embodiments of the invention employing the concept illustrated in FIG. **1C** in skew swept-back and swept-forward constructions, respectively;

FIG. 4 is a fragmentary isometric view illustrating shaping of the shroud in accordance with the invention to be compatible with the natural streamline deflection due to sweep on the suction side of the blade.

In FIG. **1A**, a swept wing, blade or foil of infinite extent is illustrated, subject to an incident supersonic flow. Since there is no spanwise variation in the wing, blade or foil geometry, the axial flow component has no aerodynamic effect. This may be clearly understood by imagining an observer who moves along the wing span at the same speed as the axial flow component. This observer will see only the normal flow component passing over the wing, i.e. he sees simply flow over an unswept wing of infinite extent. Thus, the aerodynamic behavior is determined by the flow component that is normal to the structure's leading edge. If the normal flow component is subsonic there are no shock waves associated with the flow over this structure, though, of course, to be completely shockless, the normal component must be sufficiently subsonic that transonic flow effects do not occur in the normal flow plane; i.e. the normal flow component must not accelerate to sonic speed as it passes over the same. The only effect of the axial component is in the structure of the viscous boundary layer on the wing, blade or foil surface, but this is not related to the presence or absence of shock waves. The cross-sectional shape, sharpness, and alignment of the leading edge region are not important factors in the elimination of shock waves. The shock-free character of the flow derives from the effect of sweep (a three-dimensional effect) and not from the airfoil section properties. The same ideas to obtain shock-free flow are applicable, of course, to an infinite span swept-back cascade of airfoils, as well.

FIG. **1B**, however, shows a finite span wing, blade or foil, swept-back to have "subsonic" leading edges; i.e. the normal component of flow to the leading edge is subsonic. The aerodynamics are now considerably more complicated. In particular, the presence of conical shocks, so-labelled, at the front and rear of the wing root and at the rear of the tips is unavoidable. These isolated points on the structure are discontinuities in the otherwise subsonic edges. The conical shocks are, however, weaker than their two-dimensional counterparts, and due to their three dimensional nature, decrease in strength with distance from their point of origin.

The application of a subsonic leading edge to a fan blade, is illustrated in FIG. **1C**. This illustration is simplified to its essential form, showing only the radial change in Mach number. The actual process is nonplanar due to the change in direction of the inflow with radial location. The particular case illustrated applies to a transonic fan since part of the relative incident flow is subsonic. While the leading edge can be made completely shockless, even though the blade is of finite extent, there may, however, be a weak conical shock arising from interference between the shroud at the pressure field at the blade tip, later discussed. The local leading edge sweep at each radial station is chosen to be greater than the Mach angle of the local flow. In the real non-planar case, it must also be verified that this sweep is sufficient to keep the local leading edge behind the Mach lines from the contiguous segment of the upstream leading edge. This assures that the normal flow to the leading edge is everywhere subsonic. Because of the gradient in Mach number, the incident flow is subsonic at the base of the blade so a shock cannot emanate from this point (unlike the wing root in FIG. **1B**). Hence the blade leading edge can be entirely shockless. If the relative inflow to the blade were completely supersonic, however, a conical shock would occur at the base of the blade. By designing the leading edge to be subsonic for the situation that produces maximum relative flow Mach number, the edge will remain subsonic under all other operating conditions.

While prior fan blades and other foils have, of course, been shaped in a myriad of ways for a variety of widely different applications and purposes as described, for example, in U.S. Pat. Nos. 2,212,041; 2,269,287; and 1,123,202; these, as will be evident from the above, have no real bearing upon the subsonic leading edge concept of the present invention or the problem underlying the same.

The weak conical shock at the tip before referred to, is caused by several sources of aerodynamic interference between the blade tip and the shroud. Secondary flow through the small clearance region between the tip and the shroud is one contributor; another is the interaction of the blade with the shroud boundary layer and vice versa. These effects occur on all fan blades and are not unique to blades with subsonic leading edges. A swept blade, however, is subject to an additional effect because the sweep intoduces a small spanwise velocity component in the flow field. This spanwise component is suppressed at the rigid shroud wall leading to the production of a pressure wave which may coalesce away from the blade to form a weak conical shock. Fortunately, this effect can be largely cancelled by proper design of the blade tip and shroud wall contours in the region of interaction. Specifically, the shroud wall should be contoured to be compatible with the natural streamline deflection due to sweep on the suction side of the blade. Because the streamline deflection is different on the two sides of the blade, the shroud **4**, FIG. **4**, can be shaped as at **4'** to be compatible with one side only. Since upstream shock waves come from the suction side, it is on this side that the

aerodynamic interference effect should be relieved. The blade tip 1' should then be shaped to correspond with the shroud contour 4'. This is illustrated schematically in FIG. 4, such shroud contouring being also employable to help reduce the more conventional sources of aerodynamic interference found on blades in general.

In summary, therefore, the subsonic leading edge concept of the invention, when properly implemented, can prevent the formation of upstream propagating shockwaves from the leading edges and the forward parts of the suction surfaces of fan blades and the like operating in subsonic relative inflow. The remaining upstream conical shock waves which occur due to aerodynamic interference of the blade tips with the shroud, and which because of their three-dimensional character, are already weaker than the essentially two-dimensional shocks found on conventional blade designs and can be greatly reduced in strength by properly contouring the shroud wall.

FIGS. 2 and 3 illustrate preferred embodiments of the invention showing the critical swept-forward and swept-backward constructions, respectively, of a plurality of circumferentially spaced similar parallel rotor fan blades adaptable to enable subsonic leading edge performance that prevents upstream shockwave and shock-related noise, the blades being rotated about an axis substantially parallel to the inflow direction. To achieve this, the blade leading edge in the intermediate region inward of the tip must be swept at an acute angle ∞ relative to the direction of the local inflow velocity vector less than the local Mach number angle of the flow (i.e. $< \sin^{-1} 1/m$) so that the velocity component normal to the blade edge (and aligned with the blade surface) is subsonic. The downstream portion of the edge must lie within the local Mach cone of the flow, with the degree by which the Mach angle exceeds the edge angle being sufficient to assure that even if the flow speed increases as it passes over the front portion of the blade, the normal component remains subsonic. Furthermore, the lines which generate the blade suction surface behind the leading edge must be similarly swept for some distance back. This distance must be sufficient to guarantee that any waves generated on the suction surface behind this point will be intercepted by the adjacent (following) blade. As an additional design refinement, the blade tip and shroud (if there is one) can be correspondingly countoured, as in FIG. 4, to reduce the aerodynamic interference due to sweep, of a swept blade, as previously described.

The actual implementation of the design may be accomplished in at least essentially two ways. First, a thin extension can be added to the supersonic portion of the leading edge of an existing blade design. This extension piece will provide a new leading edge of the proper shape. The extension can employ the features of backward and/or forward sweep, before mentioned. Alternatively, it might be applied to only that part of the supersonic portion of the leading edge which is deemed most important to render shock free. Since the extension is thin, it can be made to have little effect on the aerodynamics of existing blade designs. In this approach, the edge is meant to be passive except for its role in getting the flow into the blade row in a shock free manner.

Alternatively, and preferably, the required sweep can be integrated into the basic aerodynamic design of the blade. In its simplest form, this would merely constitute

a refinement to the extension idea just discussed; essentially conventional aerodynamics would be employed. Taken further, the sweep would be incorporated into the basic blade aerodynamics. The swept leading edge skew region could perform a more active role in turning and compressing the flow rather than only providing shock-free entry into the blade row. Because the concept does not require the leading edge to be extremely sharp or the flow alignment to be especially precise, considerable design flexibility exists. Furthermore, the subsonic leading edge criterion is not especially restrictive; i.e. the leading edge curve is not uniquely defined. Thus, there is considerable freedom when stacking blade sections to minimize structural loads. To optimize aerodynamic performance, the blade twist distribution should be altered slightly to balance radial pressure gradients produced by sweep. A good design should use an edge shape which contains the shocks even in off-design operation. Also as before, depending on the design constraints and relative importance of effects, only part of the blade supersonic region may be swept, if desired.

Returning to FIGS. 2 and 3, the swept-back and swept-forward blades are desirable acoustically since no upstream shocks are generated anywhere along the leading edge region.

The blade 1 of FIG. 2 is shown with its intermediate region 1' extending to the tip 1'', where supersonic inflow occurs in the direction of the arrow solabeled, swept-back at an angle α just less than about $\sin^{-1} 1/M$, where M is the Mach number associated with the Mach cone, also so-labelled and behind which the swept edge is maintained. Its direction of sweep, however, may encourage some radial flow as the fluid passes through the blade row, though this can be reduced or eliminated by design refinements. If the sweep-back construction requires a substantial rearward shift of blade section centers, large bending stresses may result. These stresses will be greatest at the trailing edge, which must be thin for aerodynamic reasons. The swept-back blade may thus be susceptible to structural constraints in some applications, if the region requiring sweep-skew is considerable.

The swept-forward intermediate region 2' to the tip 2'' of the blade 2 of FIG. 3, however, may have favorable steady aerodynamic performance; perhaps an improvement over unswept blades in some respects (e.g. the suppression of radial flow tendencies). If the extent of sweep is considerable, then it may also be subject to very high bending and torsional stresses.

Which of these preferred embodiments is preferable for a particular type of fan stage depends strongly on the performance requirements and other design constraints. For blades with rather limited regions of supersonic flow, either of the swept-back or swept-forward designs is appropriate. The specific choice depends on the severity of structural constraints and the desire to implement a particular aerodynamic design.

Other configurations besides those of FIGS. 2 and 3 are of course possible, and any number of blade configurations with only part of the supersonic region swept can be conceived. While these involve an additional acoustic penalty, they may have aerodynamic or structural advantages that make them acceptable or even desirable for certain applications. As before explained, moreover, the principles of the invention are applicable to other blade, wing or foil structures than the particu-

lar illustrative rotor fans, all being "blades" in the sense herein employed.

Further modifications will also occur to others skilled in this art and are considered to fall within the spirit and scope of the invention as defined in the appended claims.

What is claimed is:

1. A rotor blade system for preventing transonic and supersonic air flow leading-edge shock and shock-related noise, said system having, in combination, rotor blade means oriented generally in the direction of air flow, means for circumferentially rotating the blade means about an axis substantially parallel to the said direction at a velocity sufficient to render the air flow relative to said blade means supersonic from a predetermined intermediate leading-edge region of the blade means to the tip edge thereof, said blade means being swept from said region to said tip edge at an acute angle with respect to said direction less than substantially $\sin^{-1} 1/M$, where M is the Mach number of the supersonic flow, and small enough to insure that the contour of the successive portions of the blade means over the region of supersonic flow lies inside the Mach cone of resultant flow velocity over the blade means thereat.

2. A rotor blade system as claimed in claim 1 and in which at least part of said intermediate region is swept forward.

3. A rotor blade system as claimed in claim 1 and in which at least part of said intermediate region is swept backward.

4. A rotor blade system as claimed in claim 1 and in which said blade means is dispersed within a surrounding shroud.

5. A rotor blade sytem as claimed in claim 4 and in which the contour of said surrounding shroud is in part shaped in the direction of said airflow to conform to the natural curvature of the streamlines on the blade suction surface as caused by the sweep of the said intermediate region near the said tip edge of the blade means.

6. A rotor blade system as claimed in claim 1 and in which the blade means comprises a plurality of circumferentially spaced substantially parallel blades each similarly intermediately swept at substantially said acute angle with a swept-forward contour.

7. A rotor blade system having, in combination, rotor blade means oriented generally in the direction of air flow and provided with a contour swept at an acute angle with respect to said direction near the tip edge thereof, whereby the streamlines on suction surfaces of the blade means are curved by the sweep of said blade means near said tip edge, and means for circumferentially rotating the blade means within a surrounding shroud about an axis substantially parallel to said direction, the contour of part of the shroud being shaped in the said direction to conform to the curvature of said streamlines near the tip edge.

8. A method of preventing leading-edge shock and shock-related noise in transonic and supersonic airflow over a blade and the like, that comprises, orienting the blade generally in the direction of airflow, rendering the airflow relative to the blade supersonic from a predetermined intermediate leading-edge region of the blade to the tip edge thereof, and sweeping the contour of said blade along at least part of the region of supersonic airflow at an acute angle to said direction less than substantially $\sin^{-1} 1/M$, where M is the Mach number of the supersonic flow, and small enough to insure that the contour of the successive portions of the blade over the region of supersonic flow lies inside the Mach cone of resultant flow velocity over the blade thereat.

9. A method as claimed in claim 8 and in which said sweeping step in effected forwardly along said direction.

10. A rotor blade system as claimed in claim 1 and in which the blade means comprises a plurality of circumferentially spaced substantially parallel blades each similarly intermediately swept at substantially said acute angle with a swept-backward contour.

11. A method as claimed in claim 8 and in which said sweeping step is effected rearwardly along said direction.

* * * * *

9

United States Patent [19]

Kurzrock et al.

[11] 4,408,957

[45] Oct. 11, 1983

[54] SUPERSONIC BLADING

[75] Inventors: John W. Kurzrock; Peter C. Tramm, both of Indianapolis, Ind.

[73] Assignee: General Motors Corporation, Detroit, Mich.

[21] Appl. No.: 227,795

[22] Filed: Feb. 22, 1972

[51] Int. Cl.[3] F01D 5/14

[52] U.S. Cl. .... 416/237; 416/223 A; 415/181

[58] Field of Search 416/237, 175, 225, 233, 416/243; 415/181; 137/15.1

[56] References Cited

U.S. PATENT DOCUMENTS

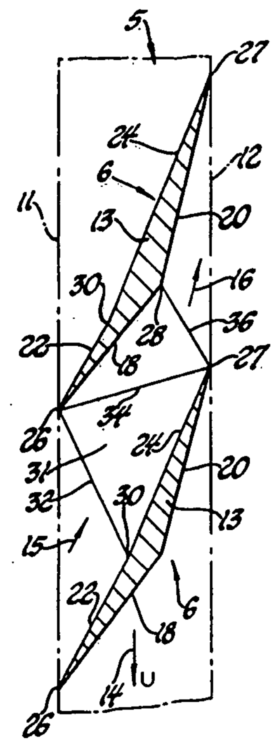
2,435,236 2/1948 Redding 416/237
2,966,028 12/1960 Johnson et al. 137/15.1
2,971,330 2/1961 Clark 137/15.1
2,974,927 3/1961 Johnson 415/199
3,059,834 10/1962 Hausammann 416/237

Primary Examiner—Stephen C. Bentley
Attorney, Agent, or Firm—Paul Fitzpatrick

[57] ABSTRACT

Supersonic rotor blading in an axial-flow fan or compressor employs blades such that shock waves originating on a blade are canceled at the intersection of the wave and the surface of an adjacent blade by virtue of a blade configuration such that the shocks intersect the blade surface at a line of properly oriented change in flow angle of the blade surface. A three-wave system is employed.

6 Claims, 5 Drawing Figures



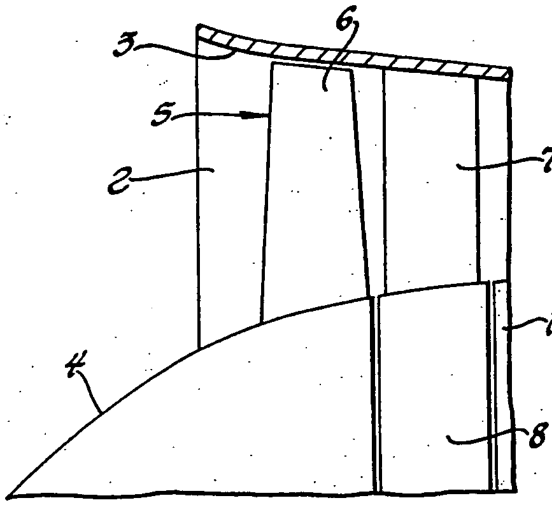


Fig. 1

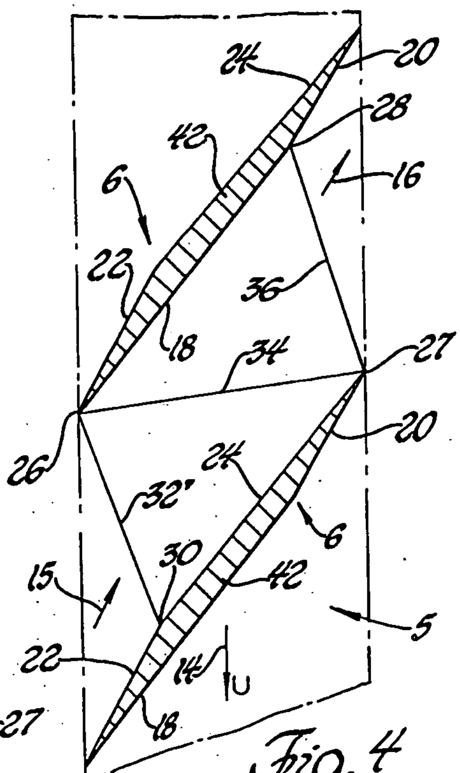


Fig. 4

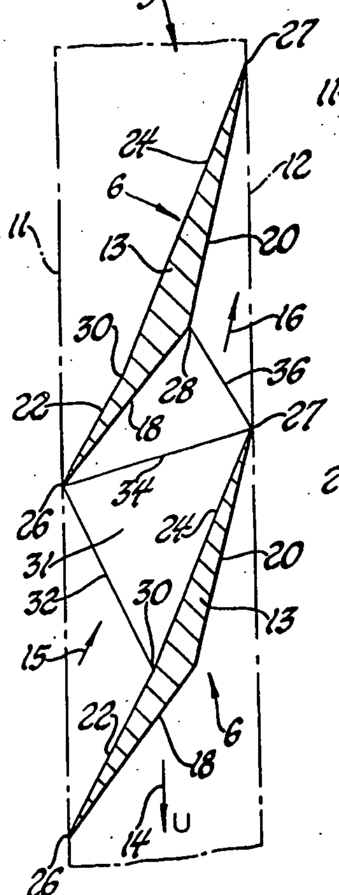


Fig. 2

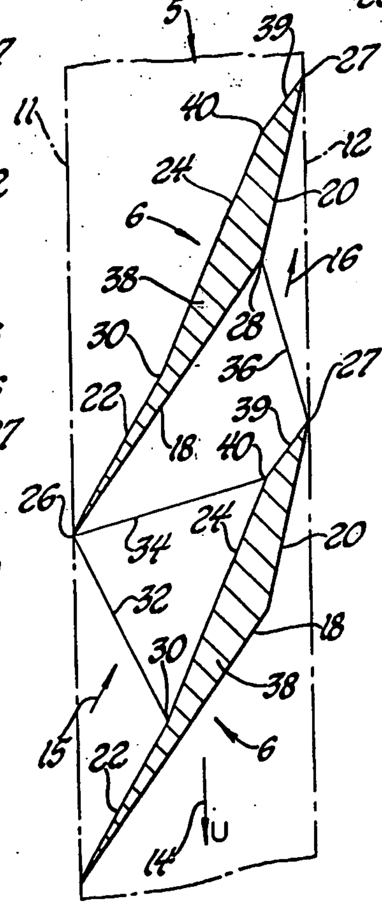


Fig. 3

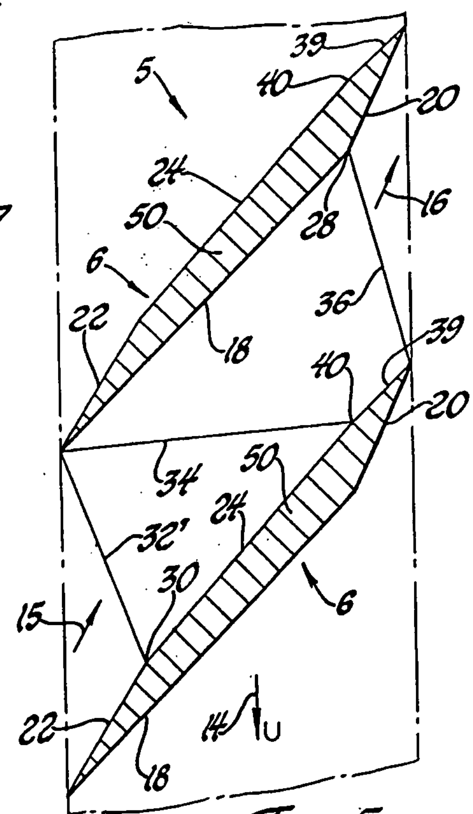


Fig. 5

SUPERSONIC BLADING

The invention described and claimed herein was made in the course of work under a contract with the Department of Defense.

Our invention is directed to supersonic blade cascades, and particularly to compressor blade cascades which form a rotor section or part of a rotor section of an axial-flow or mixed-flow compressor or fan.

The principal object of our invention is to improve the efficiency and the uniformity of exit conditions of a supersonic fan rotor or other supersonic blade cascade. An object is to obtain a predetermined static pressure ratio and exit flow angle by the use of a three-wave system.

The basic principle of our invention lies in so configuring the blades that compression or expansion waves originating on a face of a blade, as well as shock waves originating at the leading or trailing edge of a blade, intersect the surface of an adjacent blade along a line of change in direction of the surface of the adjacent blade such that the incident wave is canceled, or at the trailing edge of the adjacent blade with the same result. Particularly, in its preferred embodiment, the invention involves a three-wave system for control of supersonic flow through a blade cascade.

It will be understood that our invention relates to supersonic flow cascades, although it may be employed in rotor stages in which flow is supersonic only over part of the blade span, the blade profile according to the invention being blended into a subsonic type of blade profile toward the roots of the rotor blades in this case.

The nature of our invention and its advantages will be clear to those skilled in the art from the succeeding detailed description of preferred embodiments of the invention and the accompanying drawings.

FIG. 1 is a schematic sectional view of a supersonic or transonic compressor inlet stage.

FIG. 2 is a diagram of a first type of blade cascade in which a compression wave is produced on the suction face of each blade and the leading edge shock wave of each blade is canceled at the trailing edge of the blade leading it.

FIG. 3 illustrates a modification of the cascade of FIG. 2 in which the leading edge shock wave is canceled at an edge or line of turning of the surface of the leading blade rather than at its trailing edge.

FIG. 4 is a diagram of a blade cascade similar to that of FIG. 2 except that an expansion wave system idealized as a discrete wave is produced on the suction face of the leading blade.

FIG. 5 is a modification of FIG. 4 in which the leading edge shock wave is canceled at an edge in advance of the trailing edge of the leading blade.

Referring first to FIG. 1, to illustrate one environment for application of the invention, there is shown schematically the entry portion of an axial-flow compressor or fan. The air entry 2 is defined between a case 3 and a nose cone or fairing 4. The fairing 4 is rotatable and forms a rotor body on which is mounted an annular row or cascade 5 of rotor blades 6. The discharge from the rotor blades flows through an annular cascade of stator vanes 7 extending from the case 3 to an annular bearing support 8 which supports the shaft (not shown) which drives the fan 4, 5 and which also provides the inner boundary of the flow path through the vane cascade 7. Such structures are well known and there is no need to enlarge upon them here. Any desired structure may be disposed downstream of vanes 7, including further rotor structure 10.

As indicated, the rotor cascade 5 forms the subject matter of our invention. Our invention is concerned only with the blade portion that has supersonic inlet relative flow and supersonic or subsonic exit relative flow. Primarily, the flow field at exit is supersonic, but the three-wave principle can generate subsonic flow by having the trailing edge shock wave a strong oblique shock wave or a normal shock wave which results in subsonic flow downstream of the third wave.

Referring to FIG. 2, the forward and rearward boundaries of the annular blade cascade 5 are indicated by the broken lines 11 and 12 at the upstream and downstream or leading edge and trailing edge, boundaries respectively. In FIG. 2, two blades 6, indicated by numeral 13 in this species, are shown in cross section, which may be considered to be a typical section through a blade the cross section of which will normally vary spanwise of the blade. In FIG. 2 and in succeeding figures the blade cross section is shown as a trapezoidal figure, that is, one bounded by straight lines. This is an approximation adopted for illustrating the principles of the invention; the straight line segments illustrated may be regarded as an idealized case. In an actual compressor, the surfaces will be curved to some extent since influences such as area contractions usually exist through compressor rotor passages. Thus, the sides of the cross section may be curved, but they meet at points of relatively abrupt changes of direction at what may be termed "edges" on the blade.

The direction of rotation of the cascade is indicated by the arrow 14 and legend U in the figures, and the direction of air flow entering and leaving the cascade relative to the rotor is indicated by the arrows 15 and 16, respectively. The blade of FIG. 2 is defined by four surfaces indicated by line segments 18, 20, 22, and 24 in the cross section. Surfaces 18 and 22 meet at the leading edge 26 of the blade, and surfaces 20 and 24 meet at the trailing edge 27 of the blade. Surfaces 18 and 20 define the pressure face of the blade; that is, the face on which the higher pressure exists, which may also be termed the leading face, since it leads in the direction of rotation. Surfaces 22 and 24 define the suction or trailing face of the blade. With respect to direction of rotation, the lower blade illustrated in FIG. 2 may be considered as leading the upper blade in the figure and the upper blade of the figure as trailing the lower blade. Obviously, in an annular rotating cascade, each blade has a blade leading it and one trailing it in the cascade.

Surfaces 18 and 20 meet at an edge 28 and surfaces 22 and 24 at an edge 30. The term "edge" is employed as a term for the meeting lines of the faces as a term of convenience, as they are areas of relatively sharp curvature as compared to the relatively flat surfaces 18, 20, 22, and 24. However, the edges may be radiused to a desired extent (a desirable feature to extend the range of off-design conditions).

Two terms may be defined at this point; the pitch of the blades at any station along the span of the blades is the distance from a corresponding part of one blade in the cascade to the corresponding point on the next blade; as, for example, from the leading edge of one blade to the leading edge of the next. The term "stagger angle" as employed here refers to the angle made by a line joining the leading and trailing edges of a blade

section with a plane containing the axis of rotation of the rotor.

The dimensions and form of the blade, the pitch, and the stagger angle must be properly related in order for the blade system to operate in accordance with the principles of our invention. These principles involve the employment of three successive waves which may be shock waves, compression waves, or expansion waves, which extend across each passage 31 between adjacent blades. Strong oblique or normal shock waves may exist for the trailing edge wave. In FIG. 2, the first of these waves is a compression wave 32 produced at the suction surface of the blade by the change in surface flow angle at the edge 30. The geometry of the cascade is so chosen that this compression wave intersects the leading edge 26 of the trailing blade. The second wave 34 is a shock wave generated by the leading edge 26 of the trailing blade as illustrated. The geometry is such that this shock wave intersects the leading blade at its trailing edge 27. This shock wave turns the air flow so that it is parallel with the pressure surface 18 of the trailing blade. The shock wave is canceled at the trailing edge 27 of the leading blade. The third wave 36 is the trailing edge shock wave of the leading blade. The geometry is such that this wave intersects the trailing blade at the edge 28 between the surfaces 18 and 20. The shock wave 36 is canceled by the turning of the blade pressure face at the edge 28 at which the wave 36 intersects this face to accord with the new direction of flow of the air. The flow field behind wave 36 is uniform, with a desired static pressure ratio and exit flow angle, this angle being indicated by arrow 16. This system of three shock waves, which is illustrated only between two blades in FIG. 2, exists between all of the blades of the cascade. The wave system as described allows great flexibility in design to obtain a wide range of static pressure ratios and exit flow angles from a blade cascade.

FIG. 3 illustrates a blade system mostly similar to that of FIG. 2 and, so far as practicable, the same reference numerals will be employed as in FIG. 2 to obviate unnecessary description. The blades 38 of FIG. 3 differ from blades 13 of FIG. 2 by the provision of a third surface 39 on the suction surface of the blade disposed between surface 24 and the trailing edge 27 and joined to surface 24 at an edge 40. The compression wave 32 is generated in the same way as in the previous example and intersects the leading edge of the next blade. The leading edge shock wave, however, instead of intersecting the trailing edge of the leading blade, intersects the leading blade at the edge 40. The turning of the surface at this edge 40 is such that the flow downstream of the wave 34 is parallel to surface 39, and thus the wave is canceled. It is also parallel to surface 18 for two-dimensional flow. The trailing edge shock wave 36 is handled as in the form previously described.

The advantage of the form of FIG. 3 in some cases is the additional blade chord or closer blade setting made possible by the intersection of the leading edge shock wave 34 ahead of the trailing edge of the leading blade rather than at its trailing edge.

Considering now FIG. 4, the blades 42 of FIG. 4 have four surfaces as in FIG. 2, but the blade is of different cross-section, the obvious difference lying in the fact that the edge 30 defines a convexity rather than a concavity. Therefore, the first wave, indicated as 32' in this case, is an expansion wave system rather than a compression wave. The expansion wave system, idealized as a discrete wave from edge 30, intersects the leading edge 26 of the trailing blade, the leading edge shock wave 34 intersects the trailing edge 27 of the leading blade, and the trailing edge shock wave 36 intersects the trailing blade at the edge 28.

The blades 50 of FIG. 5 differ from blades 38 of FIG. 3 in much the same way as the blades of FIG. 4 differ from those of FIG. 2, Here, as in FIG. 3, the blades have five surfaces 18, 20, 22, 24, and 39, but these are differently arranged particularly in that, as in FIG. 4, the edge 30 is convex so that the wave 32' is an expansion wave as in FIG. 4. Wave 34 intersects the edge 40 as in FIG. 3.

As to all forms illustrated, there are two constraints involved. First, the sum of the turning angles of the successive waves must equal the desired turning angle through the cascade. Second, the desired compression of the cascade is composed of the sum of the individual waves.

If particular characteristics of any one wave are adopted, these two constraints establish the characteristics of the two remaining waves.

We do not provide specific examples of dimensions, since the blade cross-section, pitch, and stagger angles must be computed for any particular condition of flow depending upon tangential velocity of the blade, the absolute velocity of the entering air, and the sonic velocity in the air. Also, corrections should be made for bow shock waves, boundary layer phenomena, and flow mixing at exit due to finite blade trailing edge thickness. Such computations are within the range of aerodynamicists skilled in handling problems of supersonic flow.

We believe that the principles and advantages of our invention and the preferred mode of implementing it will be clear to those skilled in the art from the foregoing.

The detailed description of the preferred embodiment of the invention for the purpose of explaining the principles thereof is not to be considered as limiting or restricting the invention, since many modifications may be made by the exercise of skill in the art.

We claim:

1. A compressor rotor blade cascade suited for supersonic gas entry conditions comprising a rotor body of circular cross-section and an annular row of blades extending from the body, the blades being disposed with a substantial stagger angle, each blade having a leading pressure face and a trailing suction face, each blade including a spanwise portion the cross-sections of which are defined by at least four sides each deviating to the extent required from a straight line, the sides meeting at edges defining relatively abrupt changes in the contour of the perimeter of the blade cross-section, two of the said faces being defined by surfaces meeting at the leading edge and two by surfaces meeting at the trailing edge, each face of the blade being defined at least by two such surfaces extending chordwise and spanwise of the blade with adjacent surfaces meeting at a lateral edge extending spanwise of the blade; the blade pitch, the stagger angle, and the cross-section of the blades being such that the wave generated at the edge next downstream from the leading edge on the suction face of each blade intersects the leading edge of the next trailing blade, the leading edge shock wave of each blade intersects an edge of the next leading blade, and the trailing edge shock wave of each blade intersects a lateral edge on the pressure face of the next trailing blade, the faces being so directed downstream from

each edge as to accord with the gas flow direction downstream of the waves and cancel the incident waves.

2. A compressor rotor blade cascade suited for supersonic gas entry conditions comprising a rotor body of circular cross-section and an annular row of blades extending from the body, the blades being disposed with a substantial stagger angle, each blade having a leading pressure face and a trailing suction face, each blade including a spanwise portion the cross-sections of which are defined by at least four sides each deviating to the extent required from a straight line, the sides meeting at edges defining relatively abrupt changes in the contour of the perimeter of the blade cross-section, two of the said faces being defined by surfaces meeting at the leading edge and two by surfaces meeting at the trailing edge, each face of the blade being defined at least by two such surfaces extending chordwise and spanwise of the blade with adjacent surfaces meeting at a lateral edge extending spanwise of the blade, the lateral edge on the suction face being concave; the blade pitch, the stagger angle, and the cross-section of the blades being such that the wave generated at the edge next downstream from the leading edge on the suction face of each blade intersects the leading edge of the next trailing blade, the leading edge shock wave of each blade intersects an edge of the next leading blade, and the trailing edge shock wave of each blade intersects a lateral edge on the pressure face of the next trailing blade, the faces being so directed downstream from each edge as to accord with the gas flow direction downstream of the waves and cancel the incident waves.

3. A compressor rotor blade cascade suited for supersonic gas entry conditions comprising a rotor body of circular cross-section and an annular row of blades extending from the body, the blades being disposed with a substantial stagger angle, each blade having a leading pressure face and a trailing suction face, each blade including a spanwise portion the cross-sections of which are defined by at least four sides each deviating to the extent required from a straight line, the sides meeting at edges defining relatively abrupt changes in the contour of the perimeter of the blade cross-section, two of the said faces being defined by surfaces meeting at the leading edge and two by surfaces meeting at the trailing edge, each face of the blade being defined at least by two such surfaces extending chordwise and spanwise of the blade with adjacent surfaces meeting at a lateral edge extending spanwise of the blade, the lateral edge on the suction face being convex; the blade pitch, the stagger angle, and the cross-section of the blades being such that the wave generated at the edge next downstream from the leading edge on the suction face of each blade intersects the leading edge of the next trailing blade, the leading edge shock wave of each blade intersects an edge of the next leading blade, and the trailing edge shock wave of each blade intersects a lateral edge on the pressure face of the next trailing blade, the faces being so directed downstream from each edge as to accord with the gas flow direction downstream of the waves and cancel the incident waves.

4. A compressor rotor blade cascade suited for supersonic gas entry conditions comprising a rotor body of circular cross-section and an annular row of blades extending from the body, the blades being disposed with a substantial stagger angle, each blade having a leading pressure face and a trailing suction face, each blade including a spanwise portion the cross-sections of which are defined by at least four sides each deviating to the extent required from a straight line, the sides meeting at edges defining relatively abrupt changes in the contour of the perimeter of the blade cross-section, two of the said faces being defined by surfaces meeting at the leading edge and two by surfaces meeting at the trailing edge, the leading face of the blade being defined by two such surfaces and the trailing face by three such surfaces, the surfaces extending chordwise and spanwise of the blade with adjacent surfaces meeting at a lateral edge extending spanwise of the blade; the blade pitch, the stagger angle, and the cross-section of the blades being such that the wave generated at the edge next downstream from the leading edge on the suction face of each blade intersects the leading edge of the next trailing blade, the leading edge shock wave of each blade intersects a lateral edge of the suction face of the next leading blade, and the trailing edge shock wave of each blade intersects a lateral edge on the pressure face of the next trailing blade, the faces being so directed downstream from each edge as to accord with the gas flow direction downstream of the waves and cancel the incident waves.

5. A cascade as defined in claim 4 in which the edge next downstream from the leading edge on the suction face of each blade is concave.

6. A cascade as defined in claim 4 in which the edge next downstream from the leading edge on the suction face of each blade is convex.

* * * * *

10

US005408826A

United States Patent [19]

Stewart et al.

[11] Patent Number: 5,408,826

[45] Date of Patent: Apr. 25, 1995

[54] GAS TURBINE ENGINE CASING CONSTRUCTION

[75] Inventors: Ian F. Stewart; Sivasubramaniam K. Sathianathan; David J. Chisholm, all of Derby, England

[73] Assignee: Rolls-Royce plc, London, England

[21] Appl. No.: 216,838

[22] Filed: Mar. 24, 1994

[30] Foreign Application Priority Data

Apr. 7, 1993 [GB] United Kingdom 9307288

[51] Int. Cl.^{6} F02K 3/02

[52] U.S. Cl. 60/226.1; 60/909; 415/9

[58] Field of Search 60/226.1, 39.091, 909; 415/9, 197

[56] References Cited

U.S. PATENT DOCUMENTS

4,397,608	8/1983	Husain et al. .	
4,425,080	1/1984	Stanton et al.	605/226.1
4,490,092	12/1984	Premont	415/9
4,500,252	2/1985	Monhardt et al. .	
4,534,698	8/1985	Tomich	415/9
4,699,567	10/1987	Stewart	415/9
4,725,334	2/1988	Brimm	415/9
4,818,176	4/1989	Huether et al.	415/9
4,902,201	2/1990	Neubert	415/197
5,273,393	12/1993	Jones et al.	415/9

FOREIGN PATENT DOCUMENTS

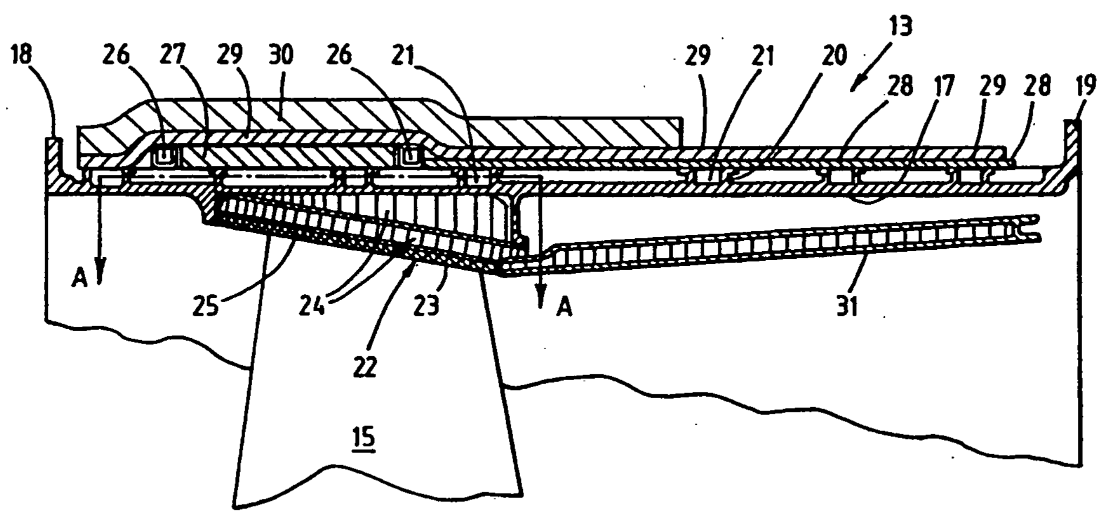
2581400	11/1986	France .	
2037900	7/1980	United Kingdom	415/9
2159886	11/1985	United Kingdom .	

Primary Examiner—Timothy S. Thorpe
Attorney, Agent, or Firm—Cushman, Darby & Cushman

[57] ABSTRACT

A casing assembly for a ducted fan gas turbine engine comprises a casing around which extend two annular, axially spaced apart, frangible rails. Pieces of cloth woven from an aromatic polyamide fibre are positioned on the casing radially outer surface between the rails. Continuous layers of the cloth are wound around the casing radially outwardly of the rails and cloth pieces. The arrangement provided enhanced containment of any fan blades of the engine which may become detached from the gas turbine engine which carries the casing assembly.

11 Claims, 2 Drawing Sheets



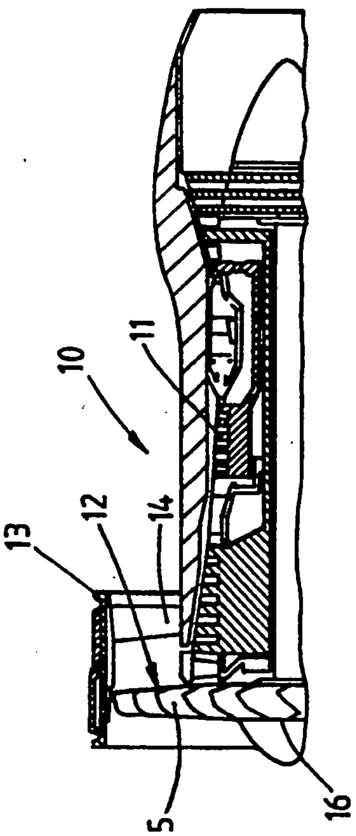


Fig.1.

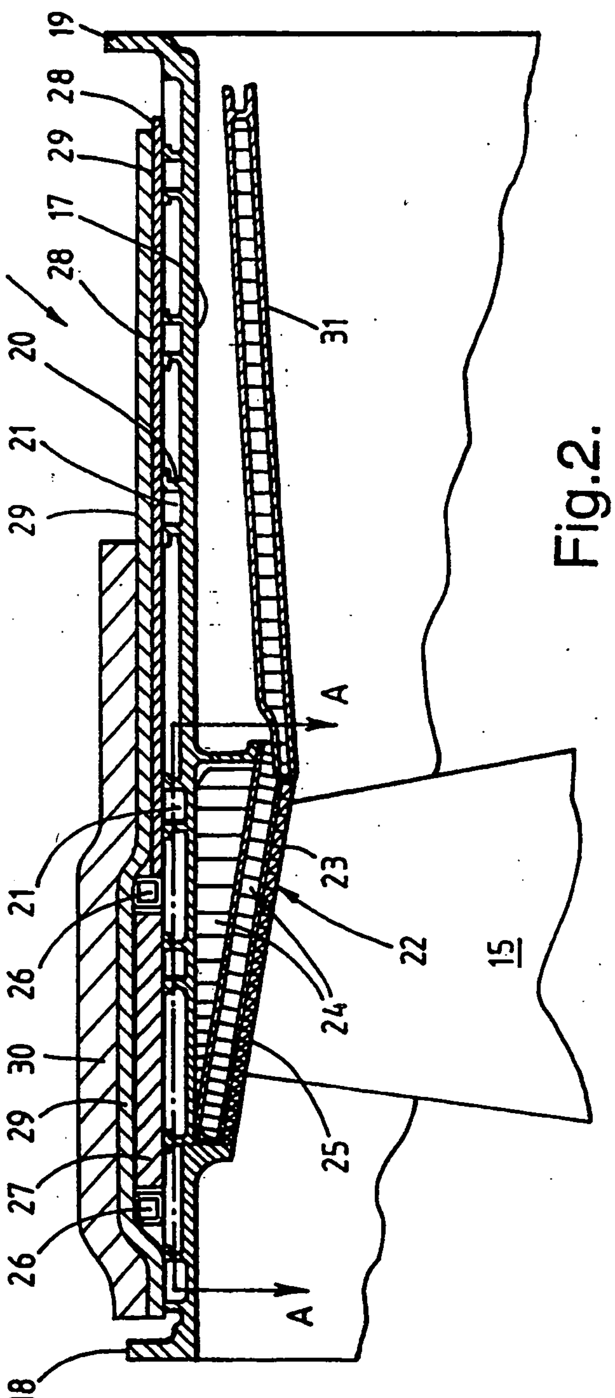


Fig.2.

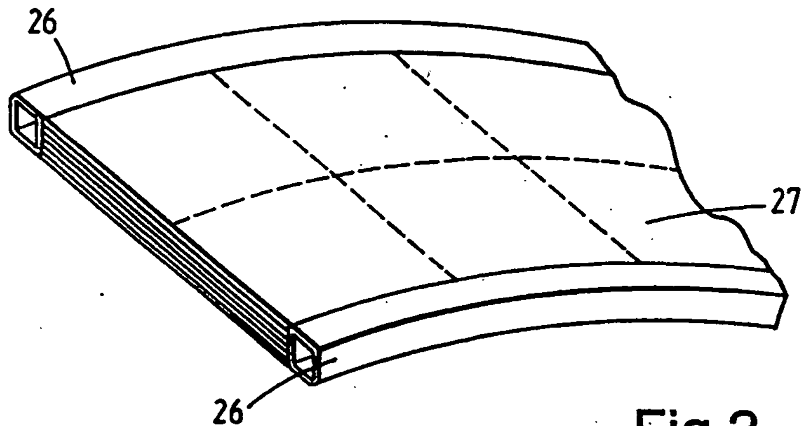


Fig.3.

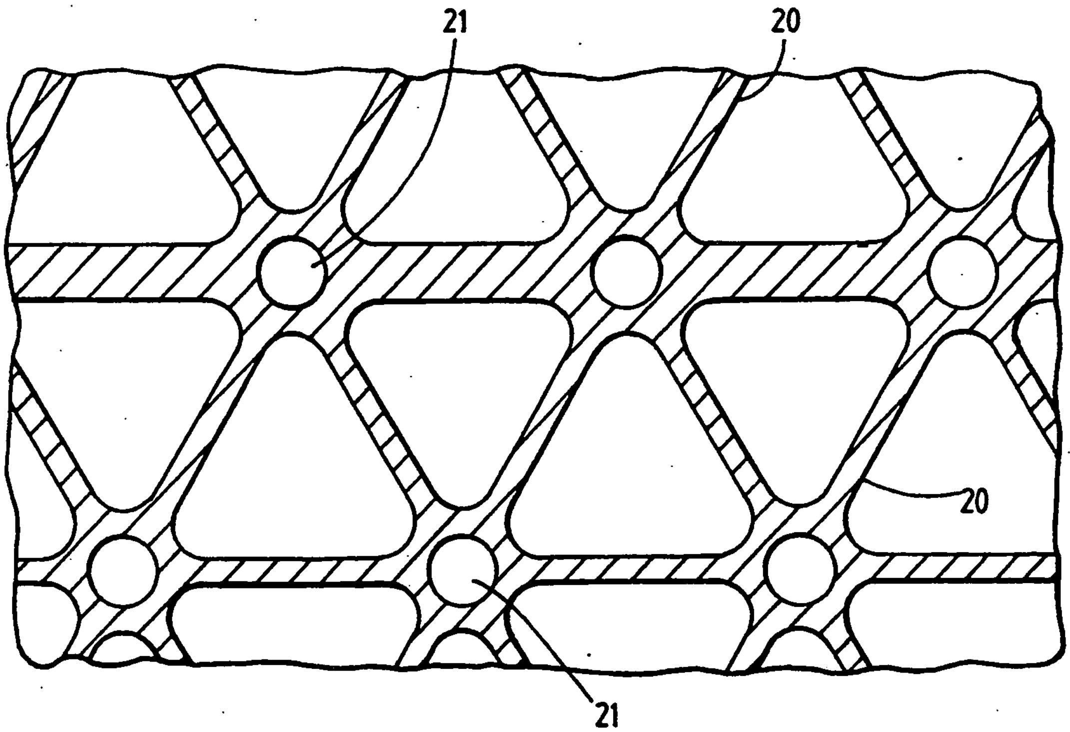


Fig.4.

GAS TURBINE ENGINE CASING CONSTRUCTION

FIELD OF THE INVENTION

This invention relates to a gas turbine engine casing construction and is particularly concerned with the construction of the fan casing of a ducted fan gas turbine engine.

BACKGROUND OF THE INVENTION

Ducted fan gas turbine engines for powering aircraft conventionally comprise a core engine which drives a propulsive fan. The fan, in turn, comprises a number of radially extending aerofoil blades mounted on a common hub and enclosed within a generally cylindrical casing.

There is a remote possibility with such engines that part or all of one or more of the fan blades could become detached from the remainder of the fan. This might be as the result of, for instance, the engine ingesting a large foreign body such as a bird. In the event of this happening, it is extremely important that the detached blade or blade portion is contained by the fan casing. Thus the fan casing must be sufficiently strong to ensure that the detached blade or blade portion does not pass through the casing and cause damage to the aircraft carrying the engine.

There are various ways in which the problem of fan blade containment may be tackled. The most obvious way is to manufacture the fan casing from an alloy which is sufficiently strong and thick to provide the desired degree of containment. However this almost invariably results in a fan casing which is undesirably heavy. An alternative approach is to provide an alloy fan casing which is thin, and therefore light, and wind around it a strong fibrous material such as an aromatic polyamide. In the event of all or part of a fan blade becoming detached, it passes through the thin alloy casing but is contained by the fibrous material.

There is a danger that the detached fan blade or blade part could cut through part of the fibre wrap, thereby reducing its ability to provide effective containment. This problem is addressed in GB2159886B by the provision of patches of the fibrous material interposed between the wound fibrous material and the casing. In the event that a fan blade or blade portion becomes detached, it pierces and passes through the casing to engage some of the patches. The patches wrap around the leading regions of the fan blade or blade portion to define a pad which in turn protects the wound fibrous material from being cut by those leading regions.

It is important for the effective operation of fan blade containment systems of this kind that the patches of fibrous material are minimally constrained. However this can be difficult to achieve since they can be compressed by the wound fibrous material which surrounds them. However if they are not tightly held by the wound material, there is a danger that they could slip into undesirable positions. Moreover any slackness in the wound material could compromise its effectiveness in providing fan blade containment.

There is a further difficulty with such fan casing constructions in that, in the interest of lightness, the alloy part of the fan casing is made as thin as possible. This can lead to a lack of adjacent stiffness in the casing. The problem is particularly severe in the case of the more powerful ducted fan engines which have very large diameter fan casings. Any thickening of the casings to provide the necessary degree stiffness is likely to increase the weight of the casings to undesirable levels.

SUMMARY OF THE INVENTION

It is an object of the present invention to provide a gas turbine engine casing construction which substantially avoids these difficulties.

According to the present invention, a gas turbine engine casing assembly comprises an annular cross-section casing configured to surround an annular array of rotary aerofoil blades, said casing defining a radially outer surface and having at least two annular, axially spaced apart, rail members on said radially outer surface and positioned coaxially therewith, discrete pieces of flexible containment material located as discontinuous layers on said casing radially outer surface between and by said rail members, and a plurality of layers of flexible containment material wound as continuous lengths around said casing radially outwardly of said rail members and of said discrete pieces.

BRIEF DESCRIPTION OF THE DRAWINGS

The present invention will now be described by way of example, with reference to the accompanying drawings in which:

FIG. 1 is a schematic sectioned side view of the upper half of a ducted fan gas turbine engine having a casing in accordance with the present invention.

FIG. 2 is a sectioned side view of part of the fan casing of the ducted fan gas turbine engine shown in FIG. 1.

FIG. 3 is a perspective view of a portion of the fan casing shown in FIG. 2.

FIG. 4 is a view on section line A—A of FIG. 2.

DETAILED DESCRIPTION OF THE INVENTION

With reference to FIG. 1, a ducted fan gas turbine engine shown at **10** is of generally conventional configuration. It comprises a core engine **11** which drives a propulsive fan **12** enclosed within a fan casing assembly **13**. The exhaust from the fan **12** is divided into two flows. The first and largest flow is directed to the exterior of the engine **10** over an annular array of outlet guide vanes **14** located at the downstream end of the fan casing assembly **13**. The outlet guide vanes **14** are generally radially extending and interconnect the fan casing assembly **13** with the core engine **11**. The remainder of the air flow from the fan **12** is directed into the core engine **11** where it is compressed and mixed with fuel before being combusted to drive the core engine **11** by conventional turbines.

The fan **12** comprises an annular array of radially extending aerofoil cross-section blades **15** mounted on a common hub **16**. During the operation of the ducted fan gas turbine engine **10**, the core engine **11** drives the fan **12** at high speed. There is a remote chance that as a result of mechanical failure, all or part of one or more of the fan blades **15** could become detached from the remainder of the fan **12**. Such mechanical failure could arise, for instance, as the result of a foreign body, such as a large bird, impacting the fan **12**. The high rotational speed of the fan **12** ensures that any such detached fan blade **15** or fan blade **15** portion is flung radially outwardly with great force towards the fan casing assembly **13**.

It is vitally important from a safety point of view that the detached fan blade **15** or fan blade **15** portion should be Contained by the fan casing assembly **13**. Thus it should not pass through the fan casing assembly **13** and cause damage to the aircraft upon which the engine **10** is mounted.

The fan casing assembly **13** is of a unique construction which ensures that any such detached fan blade **15** or fan blade **15** portion is contained by the casing assembly **13**. The construction of the casing assembly **13** can be seen more clearly if reference is now made to FIG. **2**.

The fan casing assembly **13** comprises an annular cross-section casing **17** which is supported from the core engine **11** by means of the outlet guide vanes **14** (omitted from FIG. **2** in the interests of clarity). Flanges **18** and **19** are respectively provided at the upstream and downstream ends of the casing **17** to facilitate attachment of the casing to the engine intake and outlet guide vanes (not shown) and to provide stiffening of the casing.

It is important that the casing **17** is as stiff as possible in order to avoid its distortion under load, but also as thin as possible in order to minimise its weight. To this end, the radially outer surface of the casing **17** is provided with integral stiffening ribs **20** as can be seen in FIG. **4**. The ribs **20** are distributed over virtually the whole of the radially outer surface of the casing **17**. Additionally they are interconnected so that they define a triangular pattern as is readily apparent from FIG. **4**. Blind radial holes **21** are provided at the points of intersection of the stiffening ribs to provide still further weight reduction.

The radially inner surface of the fan casing 17 supports an annular liner 22 which surrounds the radially outer extents of the fan blades **15**. The liner **22** protrudes a significant distance radially inwardly so that it terminates immediately adjacent the radially outer tips **23** of the fan blades **15**. The liner **22** also supports an annular flow defining structure **31**. The majority Of the liner **22** is formed from a metallic honeycomb material **24**, part of which is axially inclined to follow the profile of the fan blade tips **23**. The radially inner surface of the liner **22** is, however, provided with a coating **25** of a suitable abradable material. As the fan blades **15** rotate during normal engine operation, their tips **23** cut a path through the abradable coating **25**. This ensures that the radial clearance between the liner **22** and the fan blade tips **23** is as small as possible, thereby minimizing efficiency damaging air leakage across the blade tips **23**.

As well as minimizing air leakage across the blade tips **23**, the liner **22** performs two further important functions. Firstly, it assists in the stiffening of the fan casing **17**. Clearly any lack of stiffeners in the fan casing **17** could result in flexing of the liner **22** and hence changes in the clearance between the liner **22** and the fan blade tips **23**.

Secondly, in the event that the whole or part of one of the fan blades **15** should become detached, the honeycomb construction of the liner **22** defines a region which the detached fan blade **15** or fan blade **15** portion can move into. This tends to minimize the possibly damaging interaction between the detached fan blade **15** or fan blade **15** portion and the remaining fan blades **15**. It also ensures that distortion of the fan casing **17**, which will inevitably occur when the detached fan blade **15** or fan blade **15** portion impacts it, will reduce the effects of the fan casing **17** coming into contact with the remaining fan blades **15** and thereby causing additional engine damage.

The fan casing **17** is of such a thickness that in the event of a detached fan blade **15** or fan blade **15** portion coming into contact with it, it is pierced. Thus although the fan casing **17** alone is not capable of containing a detached fan blade **15** or fan blade **15** portion, it does absorb some of the kinetic energy of that blade **15**.

Containment of a detached fan blade **15** or fan blade **15** portion is provided by containment material which is provided around the radially outer surface of the fan casing **17**. More specifically, the portion of the radially outer surface of the fan casing **17** which is radially outwardly of the fan blade tips **23**, and slightly upstream thereof, is provided with two annular axially spaced apart frangible rail members **26**. The rail members **26**, which can also be seen in FIG. **3**, are hollow and of generally rectangular cross-sectional configuration. They are attached to the fan casing **17** by brazing or other suitable method and are coaxial with the fan casing **17**, thereby providing additional stiffening of the casing **17**.

The axial space between the rails **26** is filled with discrete pieces of flexible containment material **27** woven from the aromatic polyamide fibres known as "Kevlar" ("Kevlar" is a registered trade mark of Du-Pont Ltd). The pieces **27**, which are approximately 150 mm square, are loosely held together by a small number of cotton stitches *27a*. The cotton stitches *27a* serve to hold the pieces **27** together as they are wound on to the fan casing **17** between the rails **26**. Thus there are several, functionally discontinuous layers of the pieces **27** between the rails **26**; the rails **26** serving to axially locate the pieces **27**.

A first continuous length **28** of woven "Kevlar" fibre is wound around the fan casing **17** between the most downstream of the rails **26** and a region upstream of the fan casing flange **19**.

A second continuous length **29** of woven "Kevlar" fibre is wound around the rails **26**, the pieces **27** and the first continuous length **28**. Finally a third continuous length **30** of woven "Kevlar" fibre is wound around the upstream half of the second continuous length **29**. The third continuous length **30** is wound around the fan casing **17** a greater number of times than the remaining lengths **28** and **29** so that it is of the greatest radial thickness.

The rails **26** ensure that the second continuous length **29** does not exert any loads on to the pieces **27** which could put those pieces **27** into compression. It is important for the effective operation of the present invention that pieces **27** are as loose as possible.

The first continuous length **28** wound directly onto the rearward portion of the fan casing **17** serves several purposes. Firstly, it ensures that the tension in the second continuous length **29** is increased in its downstream regions. Secondly it thereby provides a reduction in the peak tensions in the second continuous length **29** as it passes over the rails **26**. Finally it provides additional blade containment which will be referred to in more detail later in the specification.

In the event that a fan blade **15** or fan blade **15** portion becomes detached from the fan **12**, it pierces the liner **22** and the fan casing **17** before encountering the pieces **27**. The pieces **27** which are impacted by the detached fan blade **15** or fan blade **15** portion effectively "blunt" the sharp edges on the blade **15** by wrapping themselves around those edges. For this to occur successfully, it is

important that each of the pieces 27 is effectively independent of its neighbouring pieces 27 and has as little inertia as possible. This is achieved by the use of the easily broken cotton threads *27a* interconnecting the woven pieces 27 and the protection afforded to the pieces 27 by the rails 26 from the constraining effect of the second continuous fibre length 29.

The detached fan blade 15 or fan blade 15 portion then encounters the second and third continuous fibre lengths 29 and 30. Since the sharp edges on the detached blade 15 are by now protected by the pieces 27, there is little likelihood that any of the continuous fibre lengths 29 and 30 will be severed by those sharp edges. Instead, the fibre lengths 29 and 30 serve to contain the detached fan blade 15 or fan blade 15 portion. This is achieved by a combination of elongation and deformation of the fibres in the second and third continuous lengths 29 and 30 and the friction between adjacent fibre layers. The second and third continuous lengths 29 and 30 thus provide containment of the detached fan blade 15 or fan blade 15 portion.

Since the rails 26 are frangible, they break up when impacted by a detached fan blade 15 or fan blade 15 portion. Consequently they do not detrimentally affect the effective blade containment provided by the second and third lengths 29 and 30. Moreover they serve to absorb some of the kinetic energy of the detached fan blade 15 or fan blade 15 portion. It will be appreciated however that although it is desirable that the rails 26 are frangible, it is not essential that they are.

There is a possibility that a detached fan blade 15 could cause additional damage to the downstream regions of the fan casing 17. For instance, while the major portion of a detached fan blade 15 will impact the casing 17 region in the vicinity of the rails 26, its radially inward or root region could impact the downstream regions of the fan casing 17. This would result in the casing 17 being pierced in this region. However the first continuous length 28 ensures that the fan blade 15 is effectively contained. It does this because although the second and third continuous lengths 29 and 30 will have been deflected by the main detached fan blade 15 impact at this point, the first continuous length 28 will not have been. By effectively containing part of the detached fan blade 15 in this manner, potentially damaging deflections of the detached fan blade 15 within the fan casing 17 are avoided, thereby limiting overall engine damage.

Although the present invention has been described with reference to the use of woven aromatic polyamide fibres for use in containment, other suitable materials may be employed if so desired. Essentially all that is necessary is that the containment material should be sufficiently strong and flexible.

Moreover, although the present invention has been described with reference to the containment of detached fan blades 15 and fan blade 15 portions, it could be applied to other parts of a gas turbine engine where there is a requirement that detached rotary parts are effectively contained.

We claim:

1. A gas turbine engine casing assembly comprising an annular cross-section casing configured to surround an annular array or rotary aerofoil blades, said casing defining a radially outer surface and having at least two annular, axially spaced apart, rail members on said radially outer surface and positioned coaxially therewith discrete pieces of flexible containment material located as discontinuous layers on said casing radially outer surface between, and by, said rail members, and a plurality of layers of flexible containment material wound as continuous lengths around said casing radially outwardly of said rail members and said discrete pieces, said rail members and said discrete pieces being completely covered by said plurality of layers, said plurality of layers being noncoextensive relative to each other.

2. A gas turbine engine casing assembly as claimed in claim 1 wherein said casing assembly is a fan casing assembly.

3. A gas turbine engine casing assembly as claimed in claim 2 wherein said plurality of layers of flexible containment material wound as continuous lengths around said casing radially outwardly of said rail members and said discrete pieces comprises two continuous extents of said material, the first of said extents being positioned radially inwardly of the second of said extents, said first extent being of greater axial extent than said second extent to extend downstream of said rails.

4. A gas turbine engine casing assembly as claimed in claim 3 wherein a third continuous extent of said flexible containment material is wound around said casing, said third continuous extent being wound around the portion of said casing downstream of said rail members radially inwardly of said second continuous extent.

5. A gas turbine engine casing assembly as claimed in claim 1 wherein said discrete pieces of flexible containment material are loosely stitched together with readily frangible threads.

6. A gas turbine engine casing assembly as claimed claim 5 wherein said readily frangible threads are of cotton.

7. A gas turbine engine casing assembly as claimed claim 1 wherein said flexible containment material comprises fabric woven from aromatic polyamide fibres.

8. A gas turbine engine casing assembly as claimed in claim 1 wherein said casing is provided on its radially outer surface with a network of integral reinforcing ribs.

9. A gas turbine engine casing assembly as claimed in claim 8 wherein said integral reinforcing ribs are in the form of inter-connected triangles.

10. A gas turbine engine casing assembly as claimed in claim 1 wherein rail members are frangible.

11. A gas turbine engine casing assembly as claimed in claim 10 wherein said frangible rail members are in the form of hollow, rectangular cross-section tubes.

* * * * *

11

United States Patent [19]

Kraig

[11] 4,012,165

[45] Mar. 15, 1977

[54] **FAN STRUCTURE**

[75] Inventor: **Alfred Henry Kraig,** Simsbury, Conn.

[73] Assignee: **United Technologies Corporation,** Hartford, Conn.

[22] Filed: **Dec. 8, 1975**

[21] Appl. No.: **638,883**

[52] **U.S. Cl.** .... **415/145;** 415/181; 137/15.1; 60/226 R

[51] **Int. Cl.**² **F01D 17/00**

[58] **Field of Search** 60/226, 662; 415/181, 415/145, 144; 137/15.1, 15.2

[56] **References Cited**

UNITED STATES PATENTS

2,623,688	12/1952	Davidson	415/181
2,628,768	2/1953	Kantrowitz	415/181
2,689,681	9/1954	Sabatiuk	415/181
2,935,246	3/1960	Roy	415/181
2,966,028	12/1960	Johnson et al.	137/15.1
3,128,939	4/1964	Szydlowski	415/181
3,203,180	8/1965	Price	60/262
3,240,012	3/1966	Price	415/145
3,396,905	8/1968	Johnson	60/226 R
3,494,539	2/1970	Littleford	60/226 R
3,528,246	9/1970	Fischer	60/226
3,549,272	12/1970	Bauger et al.	416/166
3,611,724	10/1971	Kutney	60/226
3,638,428	2/1972	Shipley et al.	60/226
3,662,556	5/1972	Poucher et al.	60/226
3,735,593	5/1973	Howell	60/226

FOREIGN PATENTS OR APPLICATIONS

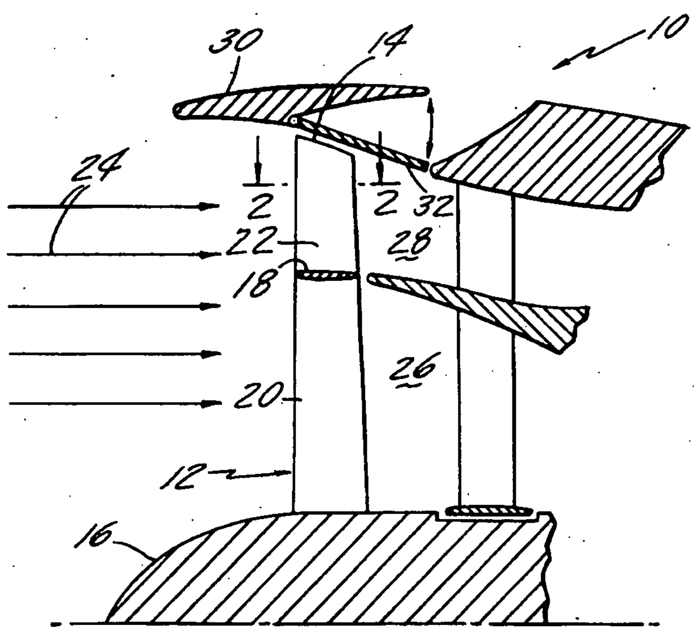
1,096,708	2/1955	France	415/145

Primary Examiner—Henry F. Raduazo
Attorney, Agent, or Firm—Robert C. Walker

[57] **ABSTRACT**

A fan structure which is particularly well suited for use in a high tip speed turbofan, gas turbine engine is disclosed. Various construction details accommodate both subsonic and supersonic flow across the fan blades. The system is built around compression shock wave phenomenon which are utilized to stably convert a portion of the dynamic energy of a supersonic stream to pressure energy within a subsonic stream.

3 Claims, 5 Drawing Figures



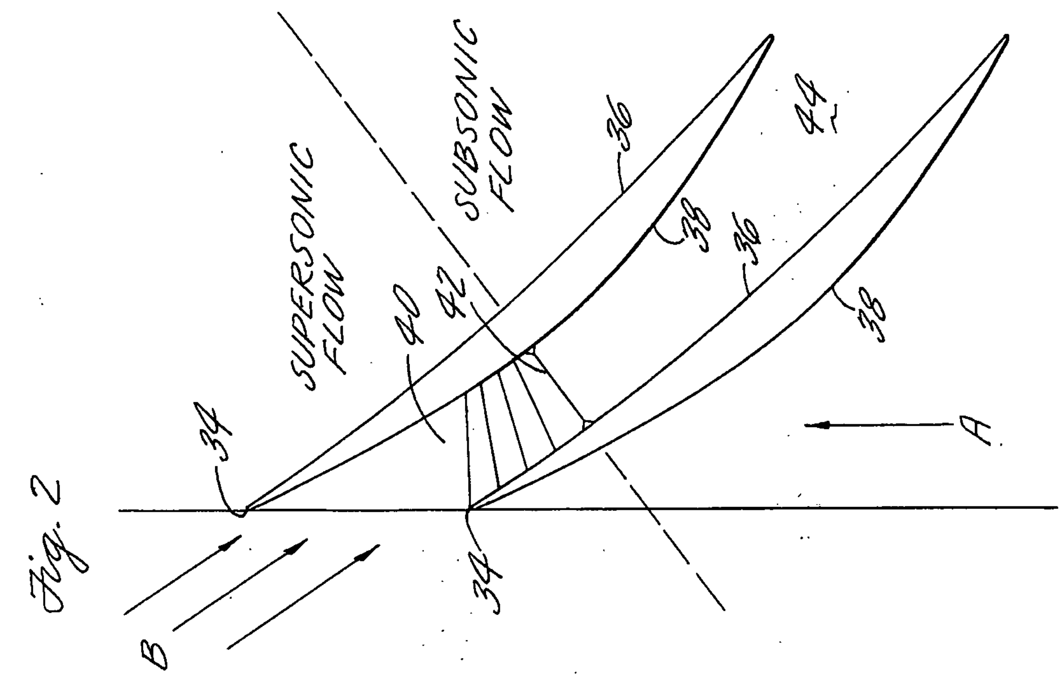


Fig. 2
SUPERSONIC FLOW
SUBSONIC FLOW
34
36
38
40
42
44
A
B

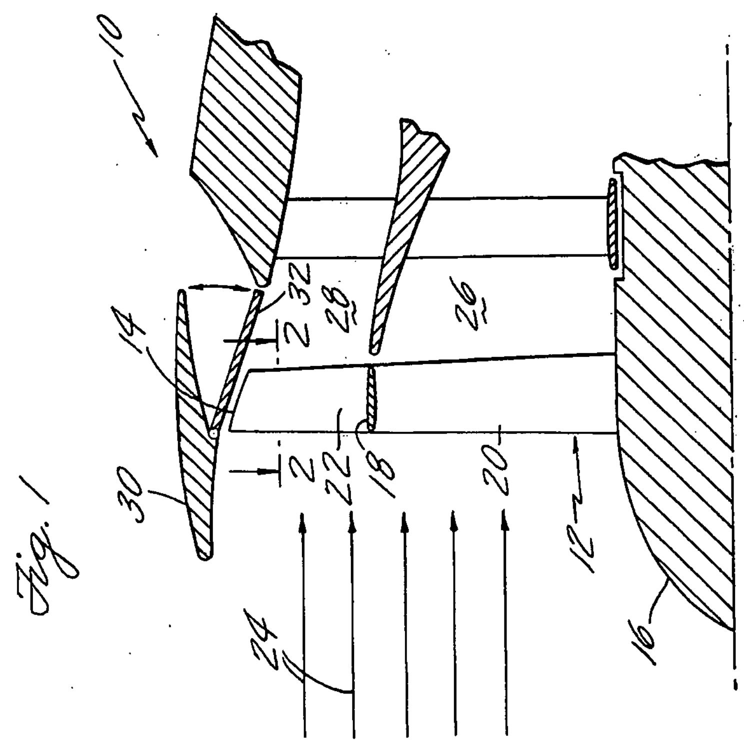


Fig. 1
10
12
14
16
18
20
22
24
26
28
30
32
2

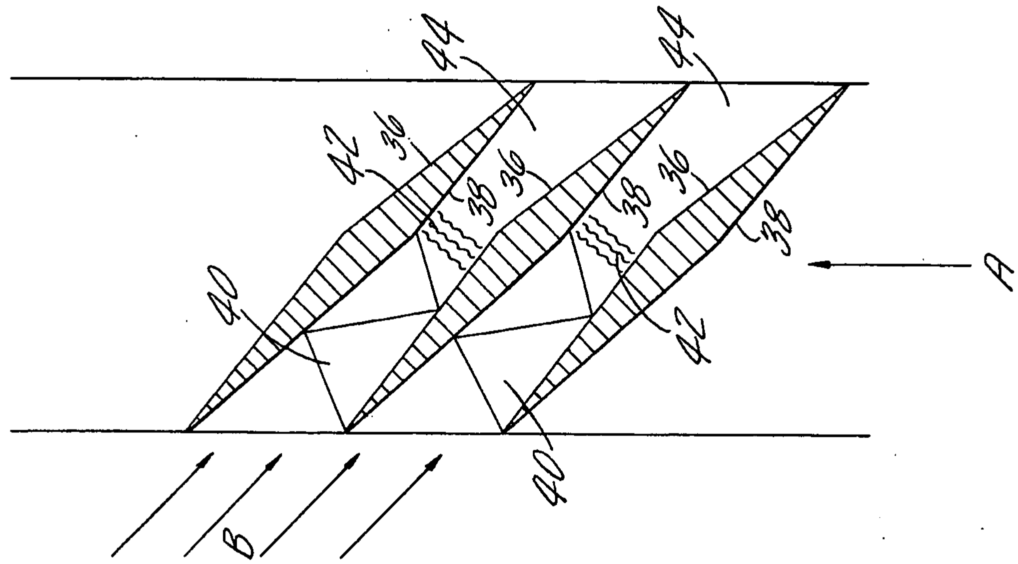


Fig. 5

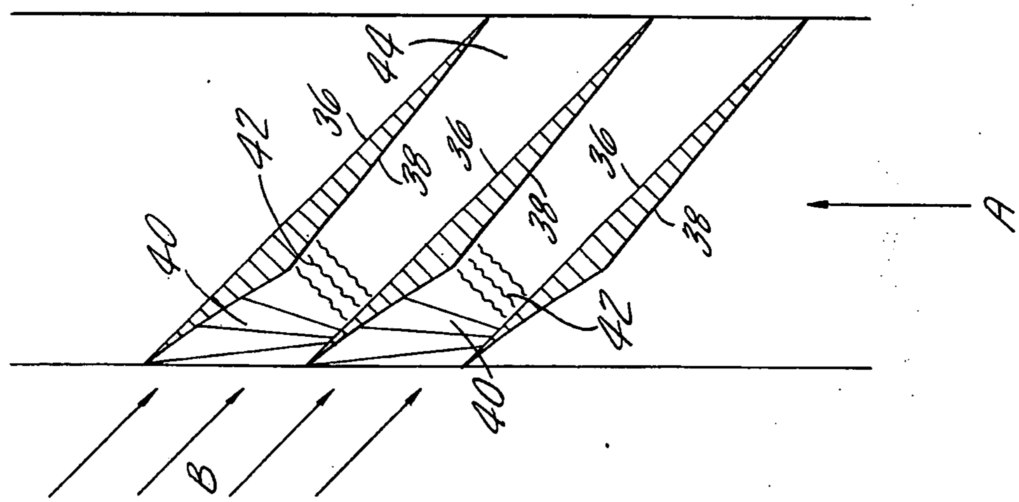


Fig. 4

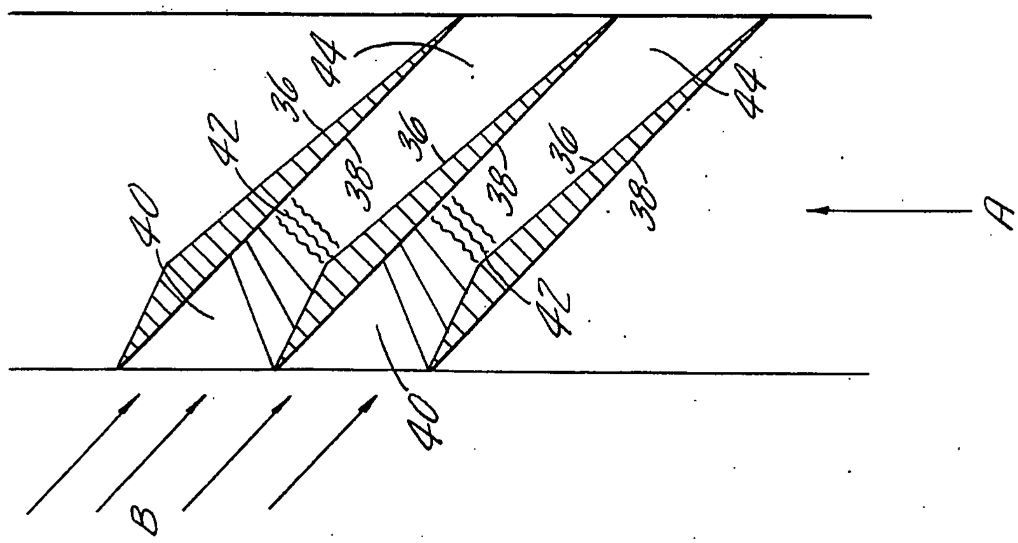


Fig. 3

FAN STRUCTURE

BACKGROUND OF THE INVENTION

1. Field of the Invention

This invention relates to turbine engines and particularly to fan engines having high blade tip speeds.

2. Description of the Prior Art

The turbofan engine is the type of power plant most widely used on large aircraft today. In a conventional turbofan engine as distinguished from a turbojet engine, a portion of the working medium gases is pumped axially through one or more compression stages and is exhausted to the atmosphere without passing through the core engine. The compression stages which exhaust to the atmosphere are called fan stages and are generally positioned at the forward end of the engine. The ratio of the air flowing through the fan stages alone to the air flowing through the core engine is referred to as the bypass ratio. The bypass ratio may be a different value for each individual engine model according to the performance requirements of that power plant. In all turbofan engines, however, the fan stages make a substantial contribution to the total engine thrust at takeoff.

For large thrust contributions a bypass ratio of 5 or greater is typical. At these bypass ratios both the diameter of the fan passage and the speed of rotation of the fan are increased from conventional, lower bypass ratio engine values to pump the working medium at a sufficient flow rate. As the diameter of the fan and the speed of fan rotation are increased, the tip speed of each individual fan blade is correspondingly increased. The formula below expresses this relationship.

$V_t = wr$

where

V_t is the velocity of the tip;

w is the angular velocity of the tip; and

r is the tip radius.

Wherever the blade speed causes the relative velocity of the approaching medium to be in excess of Mach 1.0, the dynamic energy of the supersonically approaching medium becomes converted to pressure energy by either gradual diffusion of the medium or by shock wave phenomenon. The term "subsonic and low supersonic" as applied throughout this specification to flows generally having relative approach velocities on the order of Mach 1.5 or less and the term "high supersonic" is applied to flows generally having relative approach velocities which are in excess of Mach 1.5.

In typical turbofan engines in commercial service today, the inward portion of each fan blade sees subsonic approach flow while the outward portion of each blade sees supersonic approach flow. In the outer portion of the blade a compression shock wave, commonly referred to as an "external wave," is established upstream of each blade leading edge. Across the external wave the velocity of the approaching medium is shocked from supersonic to subsonic regimes at a substantial energy loss. The greater the velocity differential across the external shock wave in an engine, the greater the loss becomes. At relative approach Mach numbers of 2 or greater, the shock wave loss is quite substantial and severe efficiency penalties are imposed against the operating engine.

A portion of the dynamic energy is converted across a shock wave to pressure energy and is recoverable in constructions capable of containing the shock wave within the rotating blades. A contained wave is most commonly referred to as an "internal wave" and has been previously proposed, by example, in U.S. Pat. No. 2,623,688 to Davidson entitled "Rotary Power Conversion Machine." Substantial aerodynamic flow losses occur in Davidson, however, as the working medium is forced through the shock wave region.

The designers of gas turbine engines are continually searching for apparatus which will reduce the severe shock wave energy losses commonly attendant in engines with high tip speed fans.

SUMMARY OF THE INVENTION

A primary aim of the present invention is to improve the operating efficiency of a turbofan, gas turbine engine having supersonic flow in the outer span region of the fan blades. A reduction in the shock wave energy losses at the outer span region is sought and, in one embodiment, a specific goal is to confine the shock wave to a constant area throat in the passage between adjacent fan blades. A reduction in aerodynamic pressure losses by controlling diffusion of the supersonically flowing medium upstream of the shock wave is a concurrent objective.

According to the present invention the flow path through the fan blades of a turbofan engine is segmented into a radially inward path for subsonic and low supersonic flow and a radially outward path for high supersonic flow, wherein the outward path includes an initial convergent region for deceleration of the approaching medium and a throat in which a normal shock wave is established during operation to convert the dynamic energy of the approaching supersonic medium to pressure energy within a subsonic stream.

A primary feature of the present invention is the segmented flow path through the fan blades. The contour of the blades in the radially inward region is conventionally adapted for subsonic and low supersonic flow and the contour of the blade in the radially outward region is adapted for high supersonic flow including a provision for a normal shock wave in the throat region thereof. A convergent zone is provided in the upstream region of the high supersonic passage between adjacent fan blades and, in one embodiment, a constant area throat region is provided immediately downstream of the convergent region. In one embodiment, starting and stabilizing doors oppose the tips of the fan blades to radially enclose the outward region. The doors are movable in response to engine operating conditions to vary the flow area of the outward region.

A primary advantage of the present invention is improved fan efficiency at high tip speeds. An internal compression geometry is incorporated within the outer region of the fan blades to increase the operating efficiency of the fan. Deceleration of the approaching medium in the upstream region of the fan blades reduces the strength of the normal shock, and correspondingly reduces the magnitude of the aerodynamic pressure losses imposed upon the flowing medium. Confinement of the shock wave to the region of minimum flow area within the passage minimizes the strength of the normal shock wave. Fan efficiency is optimized in one embodiment by starting and stabilizing doors which enable the confinement of the shock wave to the constant area throat.

The foregoing, and other objects, features and advantages of the present invention will become more apparent in the light of the following detailed description of

the preferred embodiments thereof as shown in the accompanying drawing.

BRIEF DESCRIPTION OF THE DRAWING

FIG. 1 is a simplified illustration of the fan section of a turbofan, gas turbine engine;

FIG. 2 is a sectional view taken along the line 2—2 as shown in FIG. 1 which illustrates the formation of a channel shock wave between adjacent fan blades;

FIG. 3 illustrates a convergent/divergent passage formed between adjacent fan blades;

FIG. 4 illustrates an alternate convergent/divergent passage formed between adjacent fan blades; and

FIG. 5 illustrates an alternate convergent/divergent flow path formed between adjacent fan blades.

DESCRIPTION OF THE PREFERRED EMBODIMENT

The fan section **10** of a turbofan, gas turbine engine is shown in the FIG. **1** simplified illustration. A fan blade **12** having a tip **14** extends radially outward from a rotor **16**. The fan blade shown is representative of a plurality of blades which extend radially from the rotor at the same axial position. A mid span shroud **18** separates the blade into a radially inward region **20** and a radially outward region **22**. The working medium **24** which approaches the fan section is divided by the shroud and is directed into a radially inward flow path **26** and a radially outward flow path **28**. A fan case **30** radially surrounds the tips **14** of the blades **12** and has incorporated therein a starting and stabilizing door **32** which opposes the tips of the blades. The door **32** forms the outer wall of the flow path **28** and is moveable to vary the cross sectional area of the outward flow path.

The FIG. **2** sectional view of the outer region **22** shows a pair of adjacent fan blades each having a leading edge **34**, a pressure wall **36**, and a suction wall **38**. A convergent passage **40** is formed at the upstream region of the blades between the suction wall of one blade and the opposing pressure wall of the adjacent blade. A throat **42** is formed downstream of the convergent passage at the location of minimum flow path area between the blades. A divergent passage **44** is located downstream of the throat. The direction of rotation A of the blades is shown and the direction of flow B of the approaching medium with respect to the rotating blades is illustrated.

During the operation of the engine an external shock wave is generated upstream of the leading edge of the fan blades at the outward flow path **28** as the local relative velocities of the approaching medium exceed Mach 1.0. As the approach velocities further increase the starting and stabilizing doors are opened to relieve a portion of the back pressure and cause the shock wave to be drawn downstream into the rotating fan blades. The rotating shock wave, sometimes referred to as an internal compression wave, is centered at the throat **42** between the adjacent blades and is held at that location by varying the position of the starting and stabilizing doors. In at least one embodiment the throat **42** comprises a constant area passage extending over a finite axial length between the blades. An increase in the length of the constant area throat improves the stability of the compression wave.

The starting and stabilizing doors are controlled by any of a number of control techniques sensing flow and/or pressure parameters. The doors shown provide a physical limit to the flow area although aerodynamic control of the flow area may be effectively implemented in some constructions.

The internal compression wave at the throat is a strong shock wave and, resultantly, produces a zone of unstable flow immediately adjacent thereto. Aerodynamic pressure losses are imposed upon the flowing medium as the medium flows through the unstable zone. Lower aerodynamic losses result as the velocity differential across the wave is diminished to reduce the strength of the shock wave. The apparatus of the present invention combines gradual diffusion of supersonic flow in the convergent passage **40** upstream of the throat **42** and internal compression wave phenomenon to obtain a maximized increase in pressure energy across the fan blades.

The general contour of the convergent passage **40** may be varied without departing from the concepts taught herein. For example, the passage **40** of one preferred construction illustrated by FIG. **5** is formed between a pressure wall **36** which converges with respect to the approaching flow and a suction wall **38** which also converges with respect to approaching flow. Weak oblique shocks produced in the converging region of the FIG. **5** apparatus gradually decelerate the working medium. As in all constructions, a strong passage shock occurs at the constant area throat **42** completing the transition from supersonic to subsonic flow and completing the conversion of supersonic dynamic energy to pressure energy. The subsonic medium is decelerated in the diverging passage **44** downstream of the throat to recover a substantial portion of the subsonic dynamic energy. Persons skilled in the art will recognize that additional airfoil contours which are capable of gradually diffusing the supersonic flow in an initial convergent region upstream of the throat are within the concepts taught therein.

Other convergent contours shown in this specification are capable of producing effective deceleration of the approaching medium. In the FIGS. **3** and **4** contours the pressure walls **36** or the suction walls **38** respectively converge upon the flowing medium in the passage **40** to produce weak shocks. Further tailoring of the walls, as illustrated by the concave initial region of the FIG. **4** pressure walls, decreases the strength of the weak shocks and may cause the flow to nearly approximate isentropic compression. The gradual deceleration of the supersonically flowing medium and a subsequent shock to subsonic velocities enables the apparatus disclosed herein to operate with reduced flow losses.

Although the invention has been shown and described with respect to a preferred embodiment thereof, it should be understood by those skilled in the art that various changes and omissions in the form and detail thereof may be made therein without departing from the spirit and the scope of the invention.

Having thus described a typical embodiment of my invention, that which I claim as new and desire to secure by Letters Patent of the United States is:

1. In a turbofan engine having a rotor which is mounted for rotation with respect to a stationary fan case, the improvement which comprises:

a plurality of circumferentially spaced fan blades extending radially from the rotor toward the fan case wherein each fan blade has

an inward region which is contoured for the conversion of dynamic energy within subsonic and low supersonic streams to pressure energy and

an outward region which is contoured for the conversion of dynamic energy within a high supersonic stream to pressure energy within a subsonic stream and wherein said contour of the outward region is adapted to form between each pair of adjacent blades

a convergent passage for the gradual deceleration of the supersonic stream and

a throat downstream of the convergent passage for containment of a normal shock wave between adjacent blades;

a mid-span shroud positioned on each blade to separate said inward subsonic and low supersonic region from said outward high supersonic region; and

at least one starting and stabilizing door in said stationary case which is radially opposes the fan blade tips and which is moveable with respect thereto to vary the area of the convergent passage and of the throat for positioning the shock wave at the throat.

2. The invention according to claim 1 wherein said throat is a substantially constant area passage.

3. The invention according to claim 1 in which said contour of the outer regions is further adapted to form a divergent passage downstream of said throat.

* * * * *

12

The aircraft Gas Turbine Engine and its operation

REPRINTED WITH REVISIONS
AUGUST 1988

Part No. P&W 182408

UNITED TECHNOLOGIES
PRATT & WHITNEY

Section I

Gas Turbine Engine Fundamentals

Introduction

To be able to understand a new field, one must first be familiar with its basic theories and terms. The study of jet engines has a language of its own. Special interpretations are placed on some physical laws, and special meanings are applied to common words. Section I explains the fundamentals that are necessary to understand the rudiments of a jet engine. Section II discusses the terms most commonly used in jet engine work.

The name 'jet engine' is slang. Such engines are more properly called gas turbine engines. Nevertheless, the two names are synonymous and will be used interchangeably throughout this book. The term 'gas turbine engine' might seem misleading, because the word 'gas' is so often used for gasoline. The name, however, means exactly what it says: a turbine engine that is operated by a gas rather than steam or water. The gas which operates the turbine is the product of the combustion that takes place when a fuel is mixed and burned with the air passing through the engine.

Background

The progress of powered flight has closely followed the development of suitable aircraft powerplants. Unlike the eternal question of the hen and the egg, there is no doubt as to which was necessary first. Without a lightweight and yet adequately powerful engine, controlled flight of sufficient distance to serve a useful purpose was not possible. Although Germany's Dr. N. A. Otto created the fourstroke internal combustion engine in 1876, it was not until twenty years later that Daimler was able to perfect the eight horsepower engine which enabled the Wolfert "Deutschland" to make the first gasoline powered dirigible flight. Wilbur and Orville Wright had to develop their own engine before they could achieve successful flight at Kitty Hawk in 1903. Later, Glenn H. Curtiss met with outstanding success due largely to the engines which he was instrumental in developing. And so it has gone, down through aviation history: larger and more efficient engines lead to larger, faster, and higher flying aircraft.

The first airplanes were powered by piston engines that turned a propeller, since that was the only means known at the time which could propel a heavier-than-air machine in level flight. Yet the use of jet propulsion to carry an air vehicle aloft precedes the piston engine by hundreds of years.

Most historians agree that the Chinese used rockets at least as early as the 13th-century to create terror in the ranks of their enemies. Not long after they discovered gun powder about 1000 AD, some inventive person probably learned that a cylinder filled with gunpowder and open at one end could dart across a surface when it was ignited; this may have led to other experiments, culminating in arrows on which these cylinders were tied, allowing the arrows to rocket into the air when they were ignited. These early Chinese experimentors were the first people to discover the principle of jet thrust, that is, that a stream of hot gasses exiting from one end of a tube could generate a 'push.' The same principle is the basis for today's jet engines.

Jet Propulsion Theory

Briefly, jet propulsion is the propelling force generated in the direction opposite to the flow of a mass of gas or liquid under pressure which is escaping through a hole or opening called a jet nozzle. The force that makes the nozzle of a fire hose spraying high-pressure water difficult to hold and causes the nozzle to flop around on the ground, if the nozzle is dropped, is due to jet propulsion. The thrust that sends a rocket shooting skyward is another example of the same thing. Whatever the form of the apparatus utilizing jet propulsion, the device is essentially a reaction engine (or motor) that operates through the practical application of the laws of motion that were first stated by the English physicist, Sir Isaac Newton, in 1687.

The first known reaction engine was built by Heron (sometimes called "Hero") of Alexandria somewhere around the year 250 B.C. A noted mathematician and inventor, Heron devised a machine he called an aeolipile. It consisted essentially of a closed vessel in the shape of a sphere into which steam under pressure was introduced. When the steam escaped from two bent tubes mounted opposite one another on the surface of the sphere, the tubes became jet nozzles. A force was created at the nozzles that caused the sphere to rotate about an axis. It is said that Heron attached a pulley, ropes, and linkages to the axle on which the sphere rotated, and used the device to pull open temple doors without the touch of human hands.

The principle used by Heron was later employed by Isaac Newton when he too built a forerunner of today's jet engines. Newton's invention was a wagon which he tried to propel by a steam jet pointing rearward from a steam boiler mounted on the wagon. Newton's idea was sound, but the vehicle was too underpowered to operate.

Jet Engines for Aircraft

Although jet-propelled aircraft did not become practical until the mid-1940s, the history of gas turbine propulsion goes back much further than that. In 1908, Rene Lorin, a French engineer, proposed using a piston engine to compress air which would then be mixed with fuel and burned to produce pulses of hot gases that would be expelled through a nozzle to generate propelling power. But it remained for Sir Frank Whittle of Great Britain and Han von Ohain in Germany to design applications of the jet propulsion principle that could be used to power an airplane.

Whittle filed his initial patent for a turbojet engine with a centrifugal compressor in January, 1929. The first Whittle engine was called the Power Jets W.1, after its manufacturer. It flew in the British Gloster G.40 on May 15, 1941. Concurrently with Whittle, von Ohain in Germany had been at work on a jet engine for aircraft. He built and ran his first demonstration engine in 1937. His first flight engine was the Hes 3B which flew in the Heinkel He 178 on August 27, 1939.

The Whittle and the von Ohain engines led to successful jet-powered fighter aircraft by the end of World War II, the most noted of which was the Messerschmitt Me 262 that was used by the German Air Force. It is interesting to note that the early English production jet engines used centrifugal compressors, whereas the production engines in Germany employed the more advanced axial-flow compressor. The pros and cons of these two compressor designs are discussed in Section III.

What is a Jet Engine?

Both the piston engine and the gas turbine develop power or thrust by burning a combustible mixture of fuel and air. Both convert the energy of the expanding gases into propulsive force. The piston engine does this by changing the energy of combustion into mechanical energy

which is used to turn a propeller. Aircraft propulsion is obtained as the propeller imparts a relatively small acceleration to a large mass of air. The gas turbine, in its basic turbojet configuration, imparts a relatively large acceleration to a smaller mass of air, and thus produces thrust or propulsive force directly. Here the similarity between the two types of engines ceases.

The simplest gas turbine engine for aircraft is a turbojet. A can-like horizontal container, which is open at both ends and is called an engine case, encloses the internal parts. The case is built in several sections to facilitate assembly and disassembly of the engine. Large quantities of air enter the front of the case. After the air is greatly heated and accelerated by being burned with fuel, the air that remains after the burning process and the gases produced by combustion are exhausted through the rear opening. A rotating compressor is located in the forward section of the engine case. The compressor is followed by a combustion chamber, burner section, or combustor, as this part of the engine is variously known. Next are the driving turbines which develop the power needed to operate the compressor. Last, there is the engine tail pipe or exhaust duct. The opening to the outside air at the rear of the pipe is called the jet nozzle.

The compressor raises the pressure and temperature of the incoming air before passing it on to the combustion chamber. Fuel is sprayed through nozzles into the front of the chamber. The resulting mixture of fuel and air is burned to produce hot, expanding gases that rush into the turbine section, causing the turbine rotors to rotate. On leaving the turbine section, the gases are expelled to the outside air through the exhaust duct (tail pipe) and the jet nozzle. The power that the turbine rotors extract from the gases is used to drive the compressor. Since the turbine rotors and the compressor are both mounted on the same shaft, they operate as a unit. Thus, it might be said that a single compressor turbojet engine has only one major moving part.

The other members of the gas turbine engine family, such as dual compressor (twin spool) engines, turbofans, turboprops, and turboshaft engines, are different versions or developments of the basic single compressor turbojet. For an explanation and illustrations of the various jet engine types, refer to Types of Jet Engines for Aircraft at the end of this section.

How a Turbojet Operates

A turbojet engine is essentially a machine designed for the sole purpose of producing high-velocity gases at the jet nozzle. The engine is started by rotating the compressor with a starter, and then igniting a mixture of fuel and air in the combustion chamber with one or more igniters which resemble automobile spark plugs. When the engine has started and its compressor is rotating at sufficient speed, the starter and the igniters are turned off. The engine will then run without further assistance as long as fuel and air in the proper proportions continue to enter the combustion chamber and burn.

The reason why a turbojet will run as it does lies in the compressor. The gases created by a fuel and air mixture burning under normal atmospheric pressure do not expand enough to do useful work. Air under pressure must be mixed with fuel before gases produced by combustion can be used to make a piston engine or turbojet engine operate. The more air that an engine can compress and use, the greater is the power or thrust that it can produce.

Finding a way to accomplish the compressing of air was the biggest challenge for designers during the early years of turbojet engine development. In England, Frank Whittle solved the problem by using a centrifugal compressor similar to those now employed in superchargers for aircraft piston engines. Whittle provided the power required to turn the compressor by mounting a gas-driven turbine immediately to the rear of an engine's combustion chamber in approximately the same manner used today.

High power is necessary to drive the compressor in a turbojet engine. Workable gas turbine engines would have been developed sooner if anyone had known how to build a turbine that would produce sufficient power to turn the compressor and yet leave enough energy in the exhaust gases to push an airplane. Superior compressor and turbine combinations eventually led to successful engines.

To indicate how much power is absorbed by a compressor of a moderately large turbojet, let us assume that we have an engine with about a 12:1 compression ratio that produces 10,000 pounds of thrust for takeoff. In this engine, the turbine has to produce approximately 35,000 shaft horsepower just to drive the compressor when the engine is operating at full thrust. Only the power left over is available to produce the thrust needed to propel the airplane.

Why Thrust, Not Horsepower?

When describing a turboprop, turboshaft, or piston engine, the accepted unit for measuring the rate of doing work is horsepower. Energy is the capacity for performing work, and power is the rate of doing work. Power is measured not by the amount of work done, but by units of accomplishment correlated with time. One horsepower is defined as 550 foot-pounds of work accomplished in one second, or 33,000 foot-pounds of work accomplished in one minute, a foot-pound being the energy required to lift one pound of weight a distance of one foot. Thus, both time and distance are necessary to compute horsepower.

$$P = \frac{F \times D}{T}$$

where: P = Power

F = Force

D = Distance

T = Time

When a turboprop or a piston engine performs work by driving a shaft that turns a propeller, torque and rpm can be used to determine the horsepower that the engine is developing. Torque is the twisting or rotary force exerted by the engine to turn the propeller, and rpm is the number of revolutions per minute of the engine crankshaft.

The terms 'power' and 'horsepower' are inappropriate for a jet because time and distance elements are not always involved. When a turbojet or turbofan is not moving forward, as in the case of an airplane standing on the ground with the engine running, the time and distance elements are zero: no movement of the engine takes place which can be measured against a period of time. Although torque and rpm are produced within the engine by the turbine, the horsepower is used entirely by the engine itself. According to the accepted equation for determining power, no power, as such, is being developed. Yet we know that the engine is pushing against its mounts and that the thrust that the engine is generating will cause the airplane to taxi forward when the aircraft brakes are released. Therefore, thrust, in pounds, not horsepower, is used as the measure of the propulsive force developed by turbojet and turbofan engines.

Thrust Horsepower

Because piston and turboshaft engines, and turboprops, deliver power through a rotating shaft, such engines can be placed in a test stand equipped with a dynamometer that uses the rpm and torque of the shaft to measure the horsepower being produced by the engine. Jet engine test stands measure engine output only in terms of pounds of thrust that cannot be conveniently converted to horsepower.

Once a jet-powered aircraft commences to move, time and distance enter the picture, and an approximate comparison can be made between the propulsive power being generated

by an engine in the aircraft and the horsepower that a piston engine might produce at the same airspeed. The equivalent of 33,000 foot-pounds per minute, or one horsepower, is 375 "mile-pounds" per hour.

The standard equation for power can be written to show that, at an airspeed of 375 miles per hour, one pound of thrust equals one horsepower. This is called thrust horsepower (THP).

$$THP = \frac{THRUST \times MPH}{375}$$

where: THP = The approximate thrust horsepower produced by a jet engine traveling at a given airspeed

Thrust = The pounds of thust being developed by the jet engine in flight

MPH = Airspeed in miles per hour

NOTE:
At 375 mph, 1 lb. of thrust = 1 thrust horsepower
At 750 mph, 1 lb. of thrust = 2 thrust horsepower

Since piston and turboprop engines, which transmit power through a shaft to drive a propeller, are usually rated in terms of brake horsepower measured by a dynamometer, the brake horsepower produced by those engines in a test stand must be multiplied by the 80-percent propeller efficiency that has been accepted as an industry standard. The product can then represent the thrust horsepower that a shaft engine with a propeller would produce at a given airspeed when the engine is compared on a thrust horsepower basis with a turbojet or turbofan in an aircraft flying at the same speed. The equation above shows how the thrust horsepower being developed by a jet engine may be calculated.

Generation of Thrust

Jet engines, rocket motors, and the propellers of piston engines develop thrust in accordance with Sir Isaac Newton's laws of motion. The first step is to understand Newton's laws.

Newton's First Law: A body at rest will remain at rest unless it is acted upon by an outside force. Example: A ball sitting on a level table will remain motionless until it is moved by some force such as a gust of wind or a push by a person's hand. A second part of Newton's First Law states that a body in motion will continue moving in a straight line at a uniform speed unless acted upon by an outside force. We are not concerned with this part of the law in jet engine work. Newton's Second Law: A change in motion is proportional to the force applied. This can be stated in another way: A force proportional to the rate of change of the velocity is produced whenever a body (or mass) is accelerated. Mathematically, Newton's Second Law may be expressed by an equation:

$F = M a$ where (in appropriate units):

F = Force
M = Mass (weight)
a = Acceleration

Example: When a person drives a nail with a hammer, the force with which the hammer hits the nail is proportional to the mass (which is proportional to the weight) of the hammer, multiplied by the amount the person accelerates the head of the hammer, from zero to final velocity. It would be difficult, for instance, to drive a large spike with a tack hammer because the head of the hammer has too little mass. Similarly, even with a large hammer, it would be a tedious task to drive a spike with only light, tapping blows because the acceleration imparted to the head of the hammer is too small.

Definition of Mass

It would be well to pause here to learn what is meant by the term, mass. Mass is a basic property of matter, being the amount of material in any body. It is called weight when the body is in a field of gravity, such as that of the Earth. However, an object far out in space, where no appreciable field of gravity exists, will retain the characteristics of mass. That is, it will still require the same force to accelerate or decelerate it, and it will require the same amount of heat to be applied to raise its temperature a specified number of degrees. Since, in the case of jet engines, we are working with mass in the Earth's field of gravity, the equation F = Ma can be rewritten to apply to a jet engine as follows:

$$F = \frac{w}{g} \times (V_2 - V_1)$$

where: F = Force in pounds

w = Flow rate in lbs/sec of air, gas, or a liquid, such as fuel

V_1 = Initial velocity of a mass of air, gas, or a liquid in ft/sec

V_2 = Final velocity of a mass of air, gas, or a liquid in ft/sec

g = Gravitational acceleration, 32.2 ft/sec/sec

NOTE: $M = \frac{w}{g}$, where M = Mass

A word of explanation is also needed regarding the acceleration due to gravity. The lower-case letter 'g' is the accepted symbol for denoting the increase every second in the speed of a falling body. Although the value for g varies slightly with geographical location and the altitude above sea level, a value of 32.2 feet per second squared is usually used for calculations that apply near the Earth's surface.

With reference to the above equation, note that the acceleration due to gravity, g, actually has two uses, the first being to obtain the velocity of a body dropping toward the Earth, attracted by the Earth's field of gravity. The second use, although related to the first, should be thought of as a weight-to-mass conversion factor when a mass (or the amount of material in a body) must be used in calculations that have no direct connection with the effects of gravity. The second use of the acceleration due to gravity applies in the equation above.

Newton's Third Law: To every action there is always an equal and opposite reaction. Example: When a horse moves a wagon, it must push against the surface of the road with its feet just as hard as it pulls on the traces to draw the wagon.

A Simple Jet Engine

All jet engines, including rocket motors, pulsejets, and ramjets (which will be explained later) belong to the class of powerplants called reaction engines. A balloon can illustrate how Newton's laws enable a turbojet (or any other type of reaction engine) to develop thrust. When the balloon is inflated with air at room temperature and the stem is held closed so no air can escape, the balloon will remain motionless on a tabletop because the air pressure inside the balloon is pressing equally on the balloon skin in all directions. No force is being exerted that will make the balloon move. When the stem of the balloon is released, the air escapes through the open stem because there is no longer a surface to hold the air inside the balloon. In addition, this removal of a small section of the balloon surface area produces unbalanced forces on the skin of the balloon by taking away the force which pressed on the area that was closed by the stem. The force applied to all the rest of the skin of the balloon remains as it was before. Consequently, the resulting pressure imbalance makes the balloon move in the direction away from the stem.

It is the internal imbalance of forces within the balloon (or gas turbine engine, rocket,

pulsejet, or ramjet) that gives all reaction engines their name. This is also the reason why reaction engines are able to produce thrust without actually pushing on the air outside their exhaust nozzles. Both the action and reaction forces described by Sir Isaac Newton occur inside the engine, rocket motor, or other reaction propulsion device. It must be noted that the toy balloon would have traveled across the room even if the room had been a large vacuum chamber. Rockets, for example, (which carry their own fuel and oxidizer) can operate in the airless realm of outer space.

The Thrust Equation for a Turbojet

Going now from the balloon to a jet engine, the propulsive force developed by a jet engine is the result of a complex series of actions and reactions that occur within the engine. Since the internal forces have as their objective the acceleration of a mass of hot gases, the F = Ma equation is used to calculate thrust when some mechanical means, such as an engine ground test stand equipped with a thrust measuring device, is not available. The equation used to calculate jet engine thrust is the same as the one previously used to explain the definition of mass and is repeated below for ready reference.

$$F = \frac{w}{g} x \left(V_2 - V_1\right)$$

NOTE 1: $M = \frac{w}{g}$; where M = Mass

NOTE 2: While each term in the expression $\left(V_2 - V_1\right)$ is a velocity in feet per second, the difference in the velocities represents an acceleration, a, in ft/sec^2. It should be noted that a change in either the mass, M, or the acceleration a, in F = Ma, will produce a change in the force, F. This point will be seen more clearly later.

The equation for determining the thrust produced by a turbojet will only be used in actual practice by an engineer. Nevertheless, an explanation of what this equation is and how it is derived will illustrate how a turbojet develops thrust.

It was stated earlier that a turbojet is primarily a machine for producing high-velocity gases. Another way to express this would be to say that a turbojet is essentially a machine that increases the momentum of the mass of air and fuel that passes through it. Momentum is the impulse imparted to the air, fuel, and products of combustion that pass through the engine. Mathematically, momentum is the product of the velocity of a body in motion multiplied by its mass. Since mass is the weight of a body in a gravitational field divided by the acceleration due to gravity, g, then momentum in the case of a turbojet is w/g x V, where, as before, w is the flow rate of the air and gas through the engine in pounds per second.

The $F = w/g \times (V_2 - V_1)$ version of the F = Ma equation (as it applies to a turbojet) can be transposed and rewritten:

$$F = \left(\frac{w}{g} x V_2\right) - \left(\frac{w}{g} x V_1\right)$$

Thus, by substituting the word 'momentum' for the parts of this equation expressed by $w/g \times V_2$ and $w/g \times V_1$, the force generated when a mass is accelerated through a jet engine becomes the difference between the final momentum of the mass and the initial momentum of the mass.

The incoming momentum of the air and the momentum of the fuel entering a turbojet must be subtracted from the outgoing momentum of the exhaust gases to arrive at the overall change in momentum, which represents the force generated within the engine. The opposite reaction to this force is, of course, the engine

thrust. The thrust developed by a turbojet results from the unbalanced forces created within the engine itself. An additional force is developed at the nozzle when the static pressure at the nozzle exit exceeds the static pressure of the ambient (outside) air. For a definition of static pressure, refer to static and total pressure under Pressure Measurement in Section II.

$$F_n = \frac{\text{OUTGOING EXHAUST}}{\text{GAS MOMENTUM}} - \left(\frac{\text{INCOMING AIR}}{\text{MOMENTUM}} + \frac{\text{INCOMING FUEL}}{\text{MOMENTUM}}\right)$$

WHERE: F_n = Net thrust in lbs.

Net Thrust

There are two kinds of thrust, net thrust and gross thrust. The thrust discussed thus far is net thrust. Gross thrust will be explained shortly. Net thrust results from the change in momentum of the mass of air and fuel that passes through the engine. Net thrust also includes the extra thrust at the jet nozzle when the static pressure at the nozzle exit exceeds the static pressure of the ambient air.

This additional thrust is present for the same reason that there was a force that moved the toy balloon. This, it will be remembered, came because there was an imbalance of pressure within the balloon after the stem was released. The imbalance between the static pressure at the nozzle of a jet engine and the static pressure of the ambient air results in the same effect. In the case of the jet, as with the balloon, the static pressure upstream (in the direction of the movement of the jet engine and the balloon) is greater than the static pressure downstream, which adds to the thrust. Actually, the way in which the additional force is generated at the nozzle of both the jet engine and the balloon is more complicated than this. The explanation has been simplified so that the basic principle can be more easily understood.

If we momentarily neglect the additional thrust generated at the jet nozzle and substitute the appropriate symbols for the words in the above equation, the equation for net thrust becomes:

$$F_n = \left(\frac{W_a + W_f}{g} \times V_j\right) - \left(\frac{W_a}{g} \times V_a + \frac{W_f}{g} \times V_f\right)$$

where:

F_n = Net thrust in lbs.
W_a = Airflow through the engine in lbs/sec
W_f = Fuel flow in lbs/sec
g = Acceleration due to gravity
V_j = Exhaust gas velocity in ft/sec
V_a = Incoming air velocity in ft/sec
V_f = Incoming fuel velocity in ft/sec

An examination of this equation will show that it is another way of expressing the jet-engine version of the F = Ma equation. As the equation now stands, the outgoing exhaust gases in the equation will have the mass, w/g, of both the air and the fuel consumed by the engine. The incoming air velocity, V_a, will be approximately equivalent to the speed of the aircraft. The incoming fuel velocity, V_f, is considered zero because the fuel is carried aboard the aircraft, and therefore will have no initial velocity relative to the engine. So the equation may be written:

$$F_n = \left(\frac{W_a + W_f}{g} \times V_j\right) - \left(\frac{W_a}{g} \times V_a + 0\right)$$

Or, transposing:

$$F_n = \frac{W_a}{g}\left(V_j - V_a\right) + \frac{W_f}{g}\left(V_j\right)$$

In the above equations, the symbols represent the same units as before except V_a = Airspeed of aircraft in ft/sec.

In most cases, particularly when conventional

subsonic jet nozzles are used, all of the pressure within the engine cannot be converted to velocity. The unconverted pressure is the pressure that represents the additional force (thrust) generated at the jet nozzle. This unconverted pressure and the thrust it generates become more pronounced as the speed of the aircraft increases. This extra thrust becomes significant at supersonic airspeeds. Hence, the net thrust equation must be written as follows:

$$F_n = \frac{\text{OUTGOING EXHAUST}}{\text{GAS MOMENTUM}} - \left(\frac{\text{INCOMING AIR}}{\text{MOMENTUM}} + \frac{\text{INCOMING FUEL}}{\text{MOMENTUM}}\right)$$

WHERE: F_n = **Net thrust in lbs.**

In practice, fuel flow is usually neglected when net thrust is computed, because the weight of air that is metered from various sections of the engine is assumed to be approximately equivalent to the weight of the fuel that is consumed. Therefore, the final equation (not considering fuel flow) for computing the net thrust produced by a turbojet engine becomes:

$$F_n = \frac{W_a}{g}\left(V_j - V_a\right) + A_j\left(P_j - P_{am}\right)$$

where:

A_j = Area of jet nozzle in sq. ft.
P_j = Static pressure at the jet nozzle discharge in lbs/sq. ft.
P_{am} = Static pressure of the ambient air at the jet nozzle in lbs/sq. ft.

Gross Thrust

Gross thrust is developed at the engine exhaust nozzle. This includes both the thrust generated by the outgoing momentum of the exhaust gases and the additional thrust resulting from the difference between the static pressure at the nozzle and the static pressure of the ambient air. Gross thrust does not take the incoming momentum of the air and fuel into consideration. Zero incoming momentum is assumed, which is true only when the engine is static. Without considering fuel flow, the equation for gross thrust is:

$$F_g = \frac{W_a}{g}(V_j) + A_j\left(P_j - P_{am}\right)$$

where the units are the same as before, except:

F_g = gross thrust in lbs

When an aircraft and engine are static, as when an engine is being run up prior to takeoff at the end of a runway, net thrust and gross thrust are equal. When the term 'thrust' is used by itself in discussing a gas turbine engine, the reference is usually to net thrust unless otherwise stated.

Where the Thrust Comes from

The foregoing discussion might cause one to think that the thrust produced by a turbojet engine is all developed at the engine jet nozzle, which is not the case. Actually, as just stated, the thrust results from the unbalanced forces and momentums created within the engine itself.

Forward thrust is produced within the engine whenever the momentum of the air or gases passing through the engine is increased. Similarly, negative or reverse thrust is created whenever the momentum of the air or gases decreases. For example, the momentum increases as air is forced through the compressor while the momentum decreases through the turbines because power is extracted from the outgoing gases. Figure 1-1 shows how the internal pressures vary throughout an engine. These pressures and the areas on which they work indicate the momentum changes within the engine. Since engine pressure is proportional to engine thrust, Figure 1-1 indicates how the overall thrust produced by the engine is developed. The final imbalance of these pressures and areas gives, as a net result, the total thrust which the engine is developing. In practice, this imbalance may be

measured or calculated in terms of pressure to enable a pilot to monitor engine thrust.

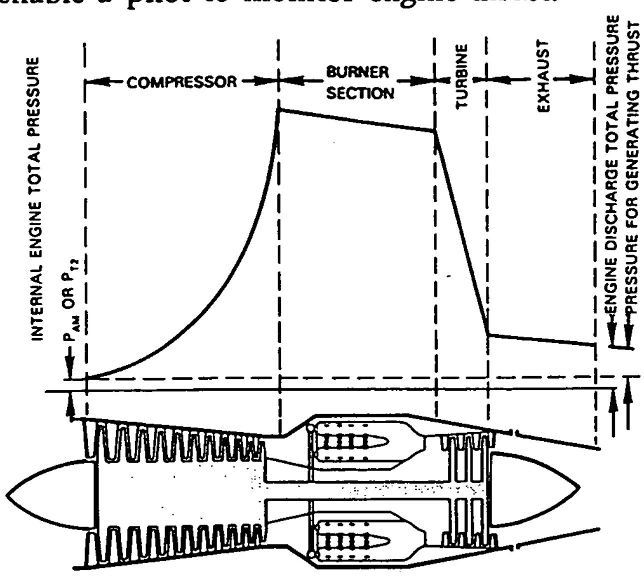


Figure 1-1 — Internal Pressure Variation inside a Typical Turbojet

Measuring Thrust in Practice

The amount of thrust that an engine produces is usually measured directly when the engine is mounted in a test stand on the ground. Normally, the engine is mounted on a floating stand which pushes against a calibrated scale. The scale has a remote indicator which permits the thrust in pounds to read directly on a dial on an instrument panel in the operator's control booth. Although thrust stands have been built to measure the static thrust of a complete aircraft and engine installation, they are seldom used because of the complications involved, and once an installed engine is airborne, the direct measurement of thrust is not practical. Therefore, for engines mounted in aircraft, some engine operating parameter (or measurable quantity) that is proportional to thrust must be used to measure the amount of thrust that an engine is producing.

On centrifugal-compressor turbojets and a few small, axial-flow compressor gas turbine engines, compressor speed in terms of rpm or percent rpm is used as a measure of thrust, because rpm is approximately proportional to the thrust being produced by the engine. For larger engines, a more precise indication of thrust is necessary. Since engine pressure and the thrust produced are proportional, most large turbojet and turbofan engines today are instrumented so that either the turbine-discharge total pressure inside the engine exhaust duct or the engine pressure ratio is shown on the pilot's instrument panel. Total pressure is the static pressure plus the rise in pressure due to the effect of ram, a term that is explained later in this section. Engine pressure ratio (EPR) is the turbine-discharge total pressure divided by the total pressure at the compressor inlet. Of the three methods for indicating thrust, EPR is the most frequently used parameter. However, engines with the opening size of their exhaust nozzles scheduled by certain engine parameters require that thrust be set by throttle position and checked by rpm, EGT, and nozzle position. How thrust is actually set by a pilot on various types of engines is explained in Section IV.

Factors Affecting Thrust

If a turbojet engine were operated only under static conditions in an air-conditioned room at standard day temperature, there would be no need to ever change the quantities used in the foregoing equations for net and gross thrust at any setting of the aircraft throttle. However, all engines installed in aircraft operate under varying conditions of airspeed and altitude. These varying conditions affect the temperature and pressure of the air entering the engine, the amount of airflow through the engine, and the gas pressure at the engine jet nozzle. This means that, for any given throttle setting, different values must be entered in the thrust equations as the airspeed and/or altitude of the aircraft changes. Although most of these variables are compensated for by the engine fuel control, many of the changes that occur affect the thrust of the engine directly. The result is a characteristic of gas turbine engines that must be understood by those who operate them. At any given throttle setting, engine thrust will vary as the temperature

and pressure of the air entering the engine changes.

A demonstration of the effect on the thrust equations of the several variables illustrates how the changing conditions at the engine air inlet affect engine performance in flight and on the ground.

Jet Nozzle Velocity

At the higher throttle settings and airspeeds encountered during normal aircraft operation, the nozzle at the rear of the engine will probably be 'choked' most of the time, which means that the gases passing through the convergent section of the nozzle will be at or near the speed of sound. As will be explained later, the speed of sound varies with air temperature, and thus, as the exhaust gas temperature varies, so also will the speed of sound. This, in turn, will affect the exhaust gas velocity at which the nozzle becomes choked. When the nozzle is choked, the only variation in the ejection velocity of the gases will be due to changes in the engine exhaust gas temperature. Velocity changes will be small under these conditions. Whenever the nozzle is not choked, varied atmospheric conditions will cause some change in jet nozzle velocity. As can be seen by the thrust equations, changes in the exhaust gas or nozzle velocity, V_j, will affect thrust. For a full explanation of the meaning of the term 'choke' refer to Subsonic, Sonic, and Supersonic Gas Flow in Section II.

Air Velocity

In an engine ground test stand, or when an aircraft is motionless on the ground, the engine must pick up air from outside the engine and accelerate this air from zero speed to the engine exhaust velocity at the jet nozzle. However, as the aircraft commences to move, the velocity of the air entering the engine also begins to increase because of the speed of the aircraft. The inlet air velocity due to airspeed appears in the equation for net thrust as airspeed, V_a. It has just been shown that the variation in jet nozzle velocity is small at the higher throttle settings because the nozzle will be choked. Therefore, the difference between V_j and V_a will become less as airspeed, or V_a, increases. In the net thrust equation, the mass flow of air is being multiplied by the difference in velocity developed through the engine. Thus, F_n is proportional to $W_a/g\ (V_j - V_a)$. When the difference in velocity decreases, the engine thrust output will also decrease. Thus, it can be seen that as aircraft speed increases, thrust decreases (Figure 1-2) for any given throttle setting when only the resulting effect on the rate of change in velocity through the engine is considered. The effect of the change in mass flow with aircraft speed will be discussed later.

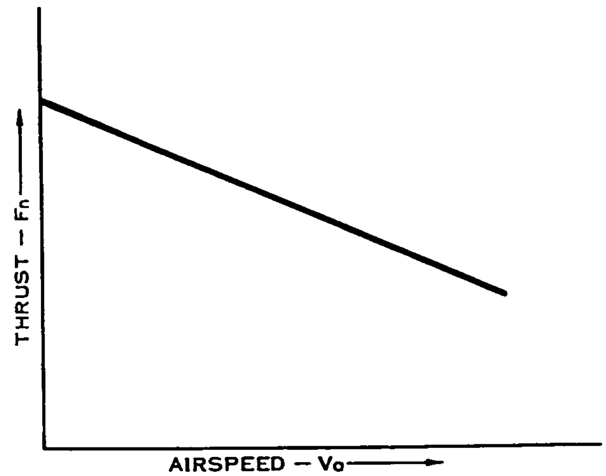


Figure 1-2 — The Effect of Airspeed on Thrust

Mass Airflow

The most significant variable in the thrust equations is mass airflow (W_a/g). Many factors affect the mass airflow, the most important being air temperature and pressure because these factors determine the density of the air entering the engine. The effect of water injection on airflow is discussed in Section III.

Air Density and the Effect of Temperature and Pressure

The density of a substance is the mass per unit of volume, or the number of molecules per cubic foot. When the density of a gas increases, there

are more molecules per cubic foot. When the density decreases, there are fewer molecules per cubic foot. The number of cubic feet of air per second that an engine consumes is controlled by the fixed engine-inlet area when the rpm is constant. Thus, unless an engine has a variable inlet area, the mass airflow into the engine at any given rpm is determined by the density of the air going into the compressor.

In free air, a rise in temperature will cause the speed of the individual molecules to increase so that they run into each other harder. When they hit each other harder, they are pushed farther apart. When they are farther apart, a given number of molecules will occupy more space, and the density of the air will be less. When a given number of molecules occupies more space, there will be fewer of them going into the engine inlet per cubic foot of air. Fewer molecules going through the engine inlet area per second means less airflow. Therefore, when the air density decreases, the airflow decreases, and less thrust will be produced by the engine. Thus, as the temperature increases, thrust decreases (Figure 1-3).

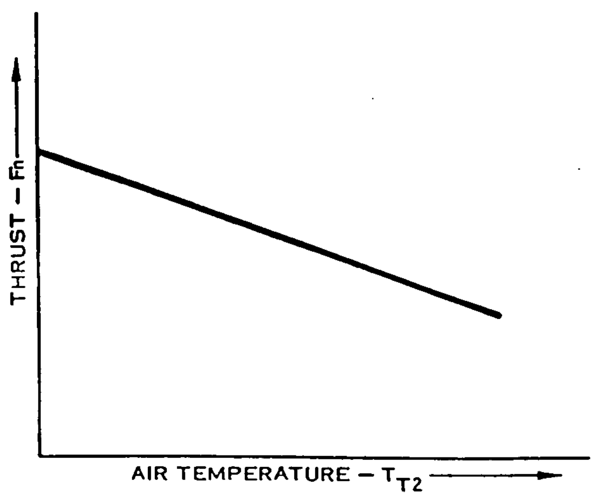


Figure 1-3 — The Effect of Air Temperature on Thrust

In free air, when the pressure increases, it is because there are more molecules per cubic foot. When there are more molecules per cubic foot, more of them will go through a fixed inlet area per given volume of air. More molecules going through the engine inlet per second means a greater airflow. As the pressure increases, the air density also increases. As air density increases the airflow increases, and, consequently, the thrust increases (Figure 1-4).

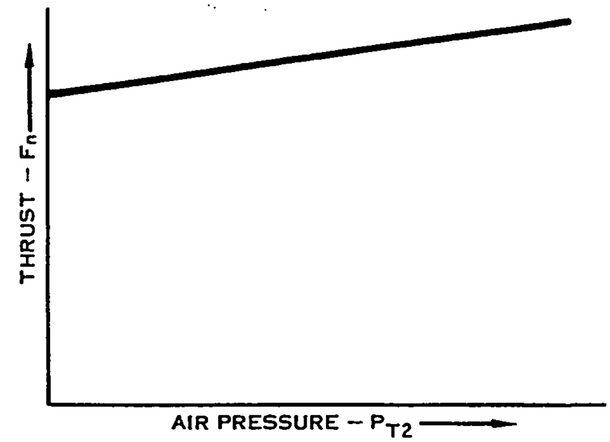


Figure 1-4 — The Effect of Air Pressure on Thrust

From this discussion it can be seen that density affects W_a in the preceding equations, and that W_a directly affects thrust. When pressure increases, as with increased airspeed or decreased altitude, density increases, and when temperature increases, as on a hot day, density decreases. Density affects thrust proportionally. A 10,000-pound-thrust engine, for instance, might generate only about 8000 pounds of thrust on a hot day, while on a cold day the same engine might produce as much as 12,000 pounds of thrust.

Altitude Effect

The effect of altitude on thrust is a function of density. As an aircraft gains altitude, the pressure of the air will lessen and the temperature of the air will become colder. As the pressure decreases, so does the thrust, but as the temperature decreases, the thrust increases. However, the pressure of the outside air decreases faster than the temperature, so an engine actually produces less thrust as the altitude is increased. As shown in Figure 2-10 of Section II, the temperature stops falling off and becomes constant at about 36,000 feet altitude, but the ambient pressure continues to drop steadily with increasing altitude. Because of this, thrust will drop off

more rapidly above 36,000 feet. If only the engine is considered, this makes 36,000 feet the best altitude for long range cruising at nominal airspeeds (Figure 1-5).

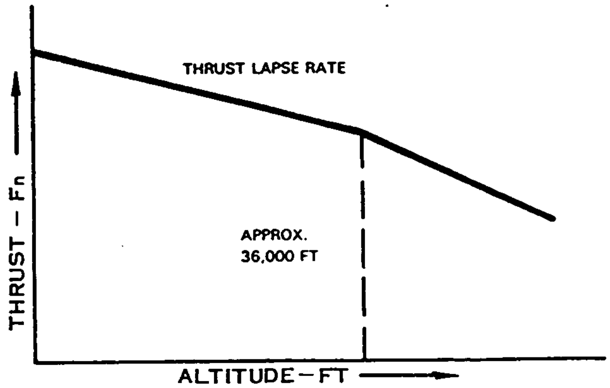


Figure 1-5 — The Effect of Altitude on Thrust

Ram Effect

The compression of air in an inlet duct arising from forward motion is called ram pressure or ram effect. As an aircraft gains speed down a runway, the outside air is moving past the aircraft with increasing speed. The effect is the same as if the aircraft were stationary in a wind tunnel and air were being blown past the aircraft by means of a fan in the tunnel. The movement of the aircraft relative to the outside air causes air to be rammed into the engine inlet duct. Ram effect increases the airflow to the engine, which, in turn, means more gross thrust.

However, ram effect alone is not all that happens at the engine air inlet as airspeed increases. There are some changes in pressure and velocity which occur inside the air inlet duct because of the shape of the duct, itself, as will be explained later. Neglecting these for the moment, it has just been shown that, as an aircraft gains airspeed, the thrust being produced by the engine decreases for any given throttle setting because V_a at the engine air inlet is increasing. Yet, because of ram effect, increasing the airspeed also increases the pressure of the air and the airflow into the engine (W_a).

What actually takes place, therefore, is the net result of these two different effects, as illustrated in Figure 1-6. In the sketch, the 'A' curve represents the tendency of thrust to drop off as airspeed builds up, due to the increase in inlet velocity, V_a. The 'B' curve represents the thrust generated by the ram effect that increases the airflow, W_a, and, consequently, increases the thrust. The 'C' curve is the result of combining curves 'A' and 'B'. Notice that the increase in thrust due to ram, as the aircraft goes faster, becomes sufficient to compensate for the loss in thrust caused by the increase in V_a. Ram will also compensate for some of the loss in thrust due to the reduced pressure at high altitude.

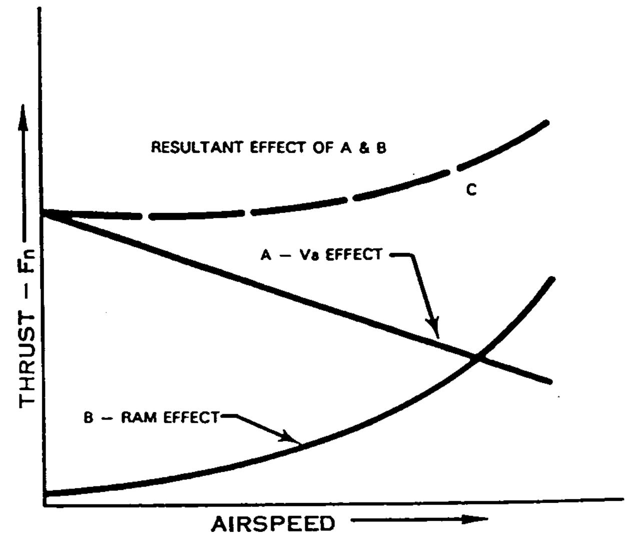


Figure 1-6 — The Effect of Ram Pressure on Thrust

Ram is important, particularly in high-speed, fighter aircraft, because eventually, when the airspeed becomes high enough, the ram effect will produce a significant overall increase in engine thrust. At the subsonic speeds at which multi-engine aircraft powered by nonafterburning engines usually cruise, ram does not affect engine thrust to the extent that occurs during supersonic flight. Nevertheless, the amount of thrust increase due to ram does become noticeable in modern subsonic jet-powered aircraft at the higher altitudes and cruise conditions.

Types of Jet Engines for Aircraft

There are several types of gas turbine powerplants for aircraft, and they are produced

in many sizes. We have already seen the principal configurations: turbojet, turbofan, turboprop, and turboshaft engines. These, in turn, can be subdivided according to the design and internal arrangement of their component parts.

Turbojet

If an aircraft gas turbine engine uses only the thrust developed within the engine to produce its propulsive force, it is a turbojet similar to the typical engine described earlier. Because they have no added features such as a fan, propeller, or free turbine, turbojets are sometimes referred to as straight jets. There are two kinds of turbojets: the centrifugal-flow compressor type and the axial-flow compressor type. Either may have one or more compressors, and some engines use both a centrifugal compressor and an axial flow compressor.

Except in Germany, the centrifugal compressor design was standard for many years because this was the type that most designers knew how to build (Figure 1-7). Centrifugal compressors operate by taking air into the engine near the compressor hub at the front of the engine. The air is then rotated at high speed by an impeller. Centrifugal force carries the air to the perimeter of the impeller and increases its total pressure. The air is then collected by a diffuser that converts the total pressure to static pressure. From the diffuser, the air is led to a manifold which feeds it to a series of individual combustion chambers, as will be shown in more detail in Section III.

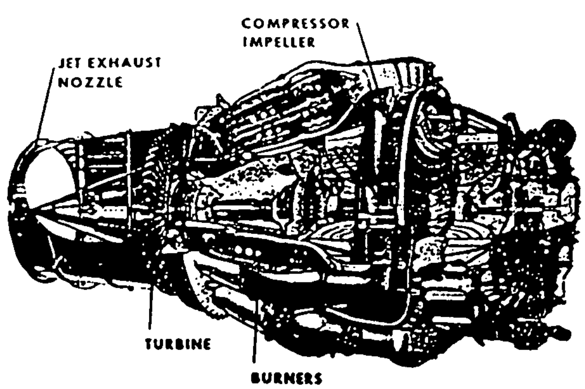


Figure 1-7 — Turbojet with a Centrifugal Compressor

The early centrifugal-compressor turbojets were both reliable and simple. Nevertheless, the amount of thrust that they can produce is limited, because the maximum compression ratio that a centrifugal compressor can attain is not very high. A centrifugal compressor is being used today in conjunction with an axial flow compressor in some very fine turboprop and turboshaft engines, an example of which is Pratt & Whitney's PT6 and ST6 engines, which will be discussed later. The improved design of these engines gives them an overall operating efficiency which is far superior to that of the early centrifugal-compressor turbojets.

Large, high performance turbojets and turbofans require the greater efficiency and higher compression ratios that can be obtained only with an axial-flow compressor. Axial-flow compressors have the added advantage of enabling an engine to have a small frontal area. Either a single compressor (Figure 1-8) or a dual

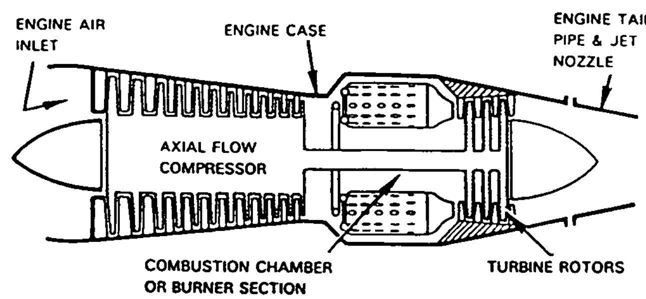


Figure 1-8 — Single -Spool Turbojet

compressor (Figure 1-9) may be used. The latter type results in very high compressor efficiency, compression ratio, and thrust. In dual compressor engines, one or more turbine stages drive the low-pressure compressor, while a separate, one- or two-stage turbine drives the high-pressure compressor. The two rotor systems operate independently of one another except for airflow. The turbine assembly for the low-pressure compressor is the rear turbine unit. This set of turbines is connected to the forward, low-pressure compressor by a shaft that passes through the hollow center of the high-pressure compressor and turbine drive shaft. The dual

compressor configuration is often called a two-spool or twin-spool engine. The single-compressor turbojet is often called a single-spool engine.

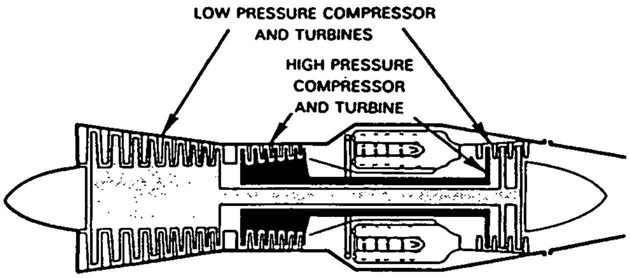


Figure 1-9 — Dual-Spool, Axial-Flow Turbojet

Because the efficiency of a turbojet is sustained at high altitude and airspeed, these engines are well suited for high-flying, high-speed aircraft that operate over a sufficient range to make the climb to their best operating altitude worthwhile. But, good as they are at their optimum altitude, high thrust at low airspeed is not a turbojet characteristic. To be at their best, turbojets need the ram air pressure at their inlet that comes only with an appreciable forward speed. Turbojet-powered aircraft therefore need long runways for takeoff.

Turboprop and Turboshaft Engines

When the exhaust gases from the basic part of a turbojet (often called a gas generator) are used to rotate an additional turbine that drives a propeller through a speed-reducing gear system, the engine becomes a turboprop (sometimes called a propjet).

In some turboprops, an extra turbine stage is incorporated in the turbine assembly that rotates the compressor. The additional power that is produced drives the propeller reduction gearing directly from the compressor drive shaft. Engines of this type are known as direct-drive turboprops (Figure 1-10). A free turbine is incorporated in most modern turboprops. This turbine is independent of the compressor-drive turbines, and is free to rotate by itself in the engine exhaust gas stream. The shaft on which the free turbine is mounted drives the propeller through the propeller reduction gear system (Figure 1-11).

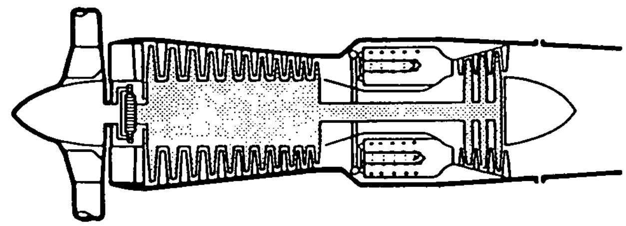


Figure 1-10 — Turboprop with Propeller Driven Directly from the Compressor Shaft through Reduction Gears

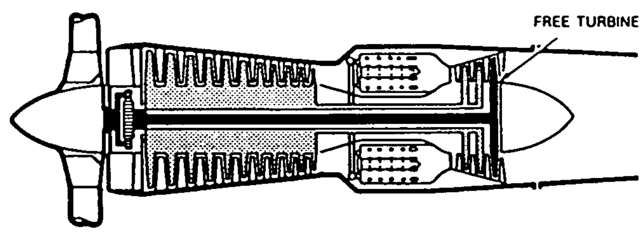


Figure 1-11 — Turboprop with Propeller Driven by a Free Turbine through Reduction Gearing

Another configuration of the free turbine turboprop has a rather unconventional rear-to-front air and gas flow direction. This configuration provides great flexibility in the design of nacelle installations: the space behind the engine is not used for an exhaust duct and can be used for wheel wells or fuel tanks. The compressor is a combination axial/centrifugal flow design. An example of such an engine is the PT6 manufactured by Pratt & Whitney Canada (Figure 1-12). The PT6 is a lightweight, free-turbine engine developing 600 to 1000 shaft horsepower, depending on the particular model.

The engine has two independent, counterrotating turbines. One turbine drives the compressor, while the other drives the propeller (in a fixed-wing aircraft) through a reduction gearbox. The compressor in the basic engine consists of three axial-flow compressor stages combined with a single centrifugal-compressor stage. The axial and centrifugal stages are assembled on the same shaft, and operate as a single unit.

Inlet air enters the engine through a circular plenum chamber near the rear of the engine, and flows forward through the successive compressor stages. The flow is directed outward by the centrifugal compressor stage through radial diffusers before entering the combustion chamber, where the flow direction is actually reversed. The gases produced by combustion are once again reversed to expand forward through each turbine stage. After leaving the turbines, the gases are collected in a peripheral exhaust scroll, and are then discharged to the atmosphere through two exhaust ports near the front of the engine.

If the shaft of a free turbine engine is used to drive something other than an aircraft propeller, such as the rotor of a helicopter, the engine is called a turboshaft or, occasionally, a shaft turbine engine. Turboshaft engines with a reduction gear system can drive boats, ships, trains, and automobiles. Frequently they are used

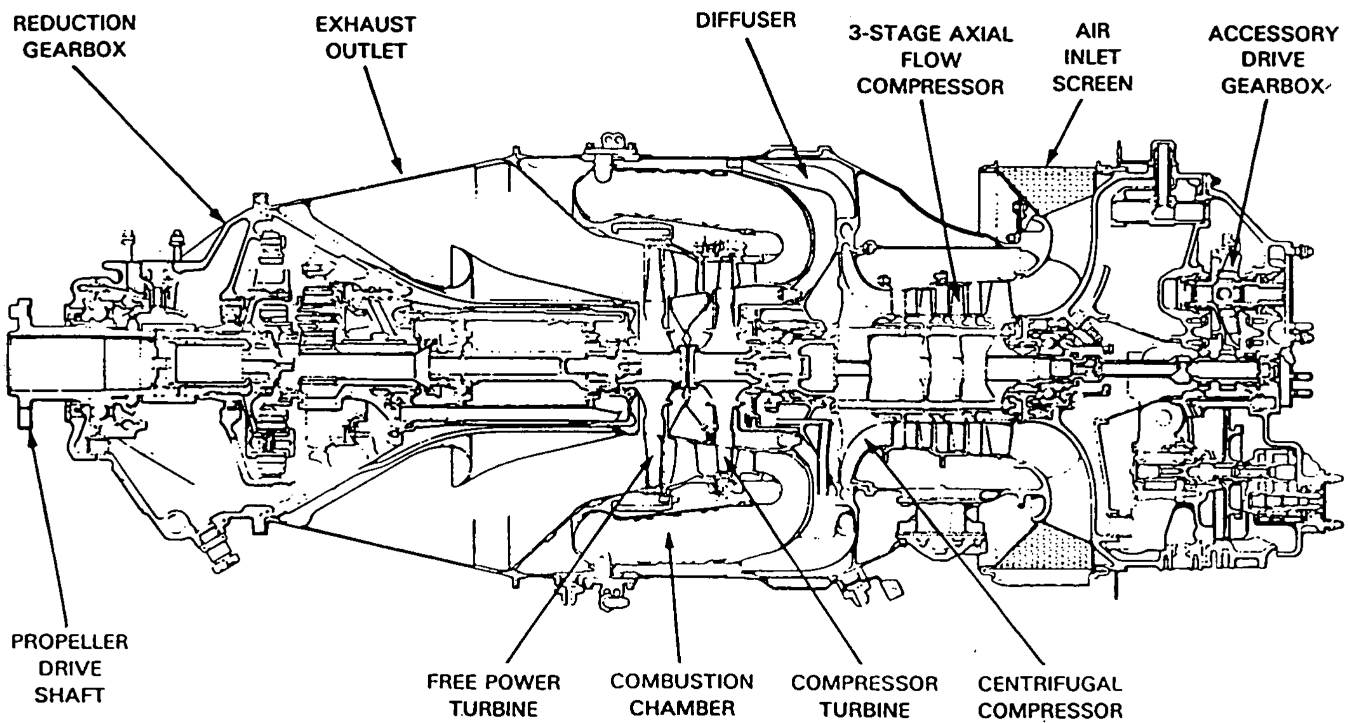


Figure 1-12 — PT6 Turboprop: a Reverse Flow, Free Turbine Design

to pump natural gas in cross-country pipelines and to drive various kinds of industrial equipment such as air compressors and large electric generators (Figure 1-13).

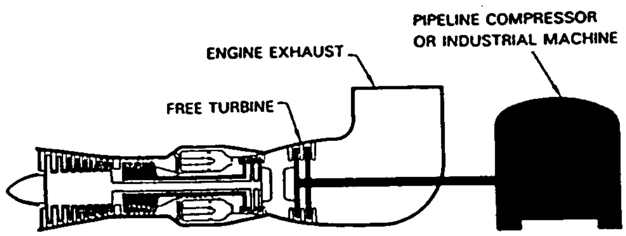


Figure 1-13 — Industrial Turboshaft Engine

Although the aircraft turboprop is more complicated and heavier than a turbojet engine of equivalent size and power, it will deliver more thrust at low subsonic airspeeds. However, the advantage decreases as flight speed increases. In normal cruising speed ranges, the propulsive efficiency (output divided by input) of a turboprop decreases as speed increases, whereas the propulsive efficiency of a turbojet increases as speed increases. The spectacular performance of a turboprop during takeoff and climb is the result of the ability of the propeller to accelerate a large mass of air while the aircraft is moving at relatively low ground and flight speed. Refer to Turbojet, Turboprop, and Turbofan Characteristics in Section V.

Turbofan

In principle, a turbofan is much like a turboprop except that the ratio of secondary airflow (the airflow through the fan or propeller) to the primary airflow through the basic engine is less. This is called the bypass ratio. Also, in the turbofan, the gear-driven propeller is replaced by an axial-flow fan with rotating blades and stationary vanes which are considerably larger but otherwise similar to the blades and vanes of an axial flow compressor. The fan is enclosed in a duct.

In a turbofan, the fan makes a substantial contribution to the total thrust. Over and above the thrust developed by the basic engine, the fan accelerates the air passing through it, similar to the function of the propeller of a turboprop. The fans of turbofan engines produce between 30 and 75 percent of the total thrust, the actual amount depending principally upon the bypass ratio.

After the secondary air leaves the fan, it does not pass through the basic engine for burning with fuel. In turbofan engines, the fan discharge air may be exhausted into the outside air through a fan jet-nozzle soon after it leaves the fan (Figure 1-14), or it may by carried rearward by an annular fan discharge duct which surrounds the basic engine for its full length (Figure 1-15). The Pratt & Whitney JT9D, PW2000, and PW4000-series turbofans are examples of engines with short ducts (Figure 1-14). The JT8D-series engines have the full-length annular duct, or long duct, as it is sometimes called (Figure 1-15).

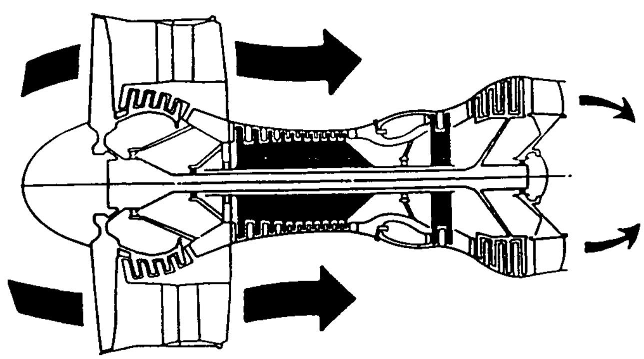


Figure 1-14 — A High Bypass Ratio Turbofan with Short Ducts

The turbofan engine depicted in Figure 1-15 has a nonmixed exhaust. That is, although the fan discharge air is carried to the rear of the engine, it is not mixed with the exhaust gases from the basic engine before being delivered to the outside air. Some long duct engines have a mixed exhaust: the fan discharge is carried to the engine tail pipe where it mixes with the exhaust from the basic engine. The air from the fan and the engine exhaust gases are then

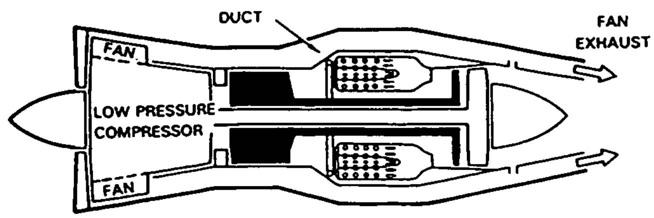


Figure 1-15 — A Dual Axial-Flow, Low Bypass Ratio Turbofan with Long Ducts and Separate Exhaust

discharged together through the engine jet nozzle.

The turbofan combines the good operating efficiency and high thrust capability of a turboprop and the high speed, high altitude capability of a turbojet. The complexity and weight of the propeller reduction gearing and the intricate propeller governing system in a turboprop is completely eliminated in a turbofan. A turbofan is therefore not only lighter than a turboprop, but is less complex.

One fundamental difference between a turbofan and a turboprop is that the airflow through the fan is controlled by the design of the engine air inlet duct in such a manner that the velocity of the air though the fan blades is not greatly affected by the speed of the aircraft. This means that the loss in propulsive efficiency with speed that becomes a limiting factor at airspeeds above 400 knots for turboprop aircraft is not a significant problem for turbofan aircraft. Turbofan engines with afterburners are used today to power supersonic airplanes that attain speeds in excess of Mach 2.0.

When compared with a turbojet of equal thrust, the turbofan has the advantage of a lower noise level for the engine exhaust, which is an important feature at all commercial airports. The lower level of noise occurs because a turbofan engine has at least one additional turbine stage to drive the fan. Extraction of more power from the engine exhaust gases as they pass through the additional turbine (or turbines) reduces the velocity of the engine exhaust. Less velocity through the jet nozzle results in less noise.

For those and other reasons (one of which is lower fuel consumption), the modern turbofan has taken over completely as the most widely used powerplant for all conventional large aircraft, both military and commercial.

Jet Engines with Afterburners

When a particular type of aircraft, such as a military fighter, needs extra bursts of speed during takeoff and climb, or for an intercept mission, each of its powerplants is usually provided with an afterburner. An engine equipped with an afterburner can develop 50 percent or more additional thrust when the afterburner is operating. Both turbojet and low bypass-ratio turbofan engines can be equipped with an afterburner, no matter what type of compressor they use. The afterburner, when added to a turbofan engine, is sometimes called an augmentor. Essentially, an afterburner (Figure 1-16 and 1-17) is simply a pipe attached to the rear of an engine instead of a tail pipe and jet nozzle. Afterburning is possible because only about 25 percent of the air (oxygen) entering the basic engine at the compressor is used to support combustion in the basic engine. The remaining air through the engine is used only for cooling.

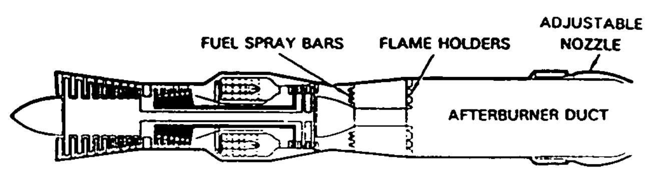


Figure 1-16 — Turbojet with Afterburner

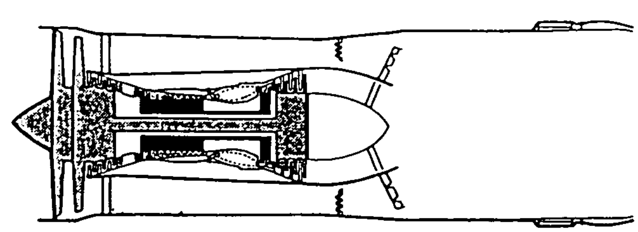


Figure 1-17 — Low Bypass Ratio Turbofan with Afterburner

In afterburning engines, fuel is injected through a fuel nozzle arrangement called 'spray bars' into the forward section of the afterburner, and is ignited. Combustion takes place because 75 percent of the air (oxygen) entering the afterburner, no longer needed for cooling, is still available for burning with fuel. As the excess air and the fuel burn together in the afterburner duct, the engine exhaust gases are again expanded by heat and accelerated. Their increased velocity provides the additional thrust. The afterburner is provided with flame holders downstream of the spray bars to prevent the flame from being blown out of the afterburner. The afterburner has an adjustable nozzle that opens automatically whenever the afterburner is on. When the afterburner is not operating, the nozzle remains in the closed (minimum opening) position on some models and continues to vary on others.

Some types of turbofan engines also use afterburning to increase thrust. In general, these engines are divided into two groups: those that use a conventional afterburning process for the mixed primary (hot) and secondary (fan) exhaust gases and airstream (Figure 1-17) and those that burn the extra fuel only in the fan air exhaust stream, which is called a duct heater. Although afterburning more than doubles the engine fuel consumption, the use of an afterburner is profitable in those aircraft that may urgently require increased speed for short periods of time or extra power for short takeoffs with heavy loads. For a more detailed discussion of afterburners, refer to Section III.

Aircraft Engines for Short Takeoff/Vertical Takeoff and Landing

Some mention should be made of engine configurations that allow fixed-wing aircraft to take off from short runways, make tighter maneuvers while in flight, or to take off and land vertically. The engines are gas turbines and operate according to the fundamentals already discussed, but their placement in the aircraft, their exhaust duct and nozzle designs, and at times their configuration, differ from the conventional jet engine.

Short takeoff and landing (STOL) capability is partly dependent on the design of an airplane's wing, which can be optimized for high lift at low runway speeds. After the wing and fuselage design, the engine is the deciding factor. Engines can provide STOL ability with a variable-angle exhaust nozzle: that is, the exhaust nozzle is flexible and can deflect the exhaust gases to help 'roll' the nose of the aircraft upward and then push the plane upward at a steep angle of climb. The U.S. Air Force's new C-17A airlifter uses an alternative method. The exhaust gases from the turbofans mounted on the wings are deflected through special 'slats' — themselves small wings — that extend from the trailing edges of the wings. The slats turn the exhaust over the back edge of the wings, which helps to accelerate the airflow over the wings, creating greater lift than would be possible by the wings themselves (Figure 1-18). Note that some STOL devices can be used while the airplane is flying to allow sharp, rapid maneuvers.

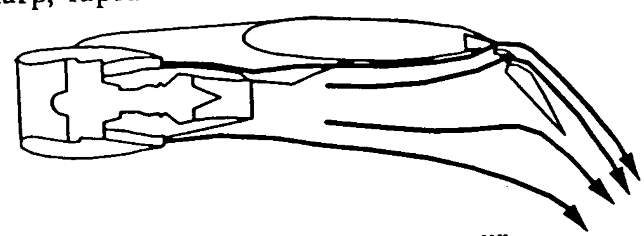


Figure 1-18 — Thrust Diverted across Wings to Enhance Wing Lift

The vertical takeoff and landing (VTOL) engine can allow STOL as well as VTOL. Depending on the need, the exhaust of the jet nozzle — or of both the jet nozzle and the fan exhaust nozzle on turbofans — can be deflected at an angle to push the aircraft at a steep climb, or the exhaust can be deflected perpendicularly to the ground to allow the airplane to take off straight upward like a helicopter (Figure 1-19). For vertical takeoff and hovering, the vectored thrust of the engines must be greater than the aircraft weight; more than 25,000 pounds of thrust would be necessary to allow a 25,000 pound aircraft to take off vertically. By contrast,

a conventional aircraft taking off from a conventional airfield requires about one quarter to one third of its weight in engine thrust. An alternative to deflecting a jet exhaust is to tilt the entire engine in the direction of takeoff. In this case the engine can be a turbojet, turbofan, or a turboprop with large-diameter propellers, the turboprop functioning most directly like a helicopter when the propeller thrust is directed straight towards the ground.

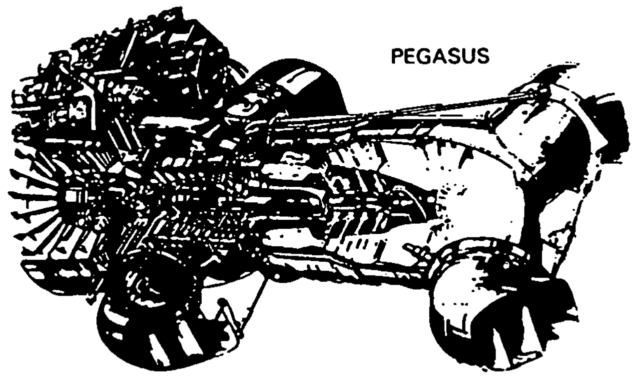


Figure 1-19 — Thrust Vectoring Nozzles for VTOL

Finally, a very different kind of VTOL engine uses a large fan built into the wing or fuselage of the aircraft. A gas generator can turn the fan mechanically by a shaft or pneumatically by pressurized air bled from the gas generator. The fan would provide vertical lift very efficiently because of its large diameter, just as turboprops and turbofans provide the best efficiency relative to turbojets. For horizontal flight, the fan could be tilted or, if it is fixed, a second set of engines designed for conventional winged flight could take over.

STOL and VTOL configurations can be designed in a variety of ways and integrated with wings and other propulsion systems. Generally, the STOL/VTOL system is optimized for certain modes of flight. For example, aircraft that must hover for extended periods might use large fans or tilted propellers for efficient hovering; aircraft that need VTOL capability only for landing and takeoff, and are expected to provide high speed and maneuverability, might use a deflected exhaust from a turbojet or turbofan. However, all of these configurations are compromises: all STOL/VTOL engines have increased weight and cost relative to ordinary engines because the thrust vectoring or engine-tilting equipment add material and complexity to the aircraft. Additionally, the transition between different flight modes may require extra devices to control the stability of the aircraft until the airflow over its wings is high enough to allow the rudder and ailerons to function; again, these extras add to the weight, cost, and complexity of the aircraft, and increase the aircraft's fuel consumption.

Propfans

The inherent efficiency of the conventional propeller has been primarily responsible for the renewed interest in its application to open-rotor engines, called 'propfans'. Propellers achieve about 80-percent propulsive efficiency whereas current high bypass-ratio turbofans deliver about 60 to 65-percent efficiency. But propeller-driven aircraft are limited to low attitudes and airspeeds, since, as aircraft speed nears 0.6 Mach number, airflow over the propellers reaches supersonic speed, which produces shockwaves that decrease propulsive efficiency. The configuration of propfan blades can overcome the limitations of conventional propellers and contribute to a decrease in fuel burn of up to 20 percent relative to high bypass-ratio turbofans.

One of the problems of large-diameter propellers is their size, which restricts the placement of the engine on the aircraft. Propfans reduce this difficulty. With more blades than conventional turboprops, propfans can have smaller diameters, which minimizes the mounting difficulties. The thinner, swept blades and larger chords of propfans also keep the critical Mach number at turbofan operating levels.

Propfans can be either single-rotation (a single propeller; Figure 1-20) or counterrotation (two propellers, one behind the other, that rotate in opposite directions; Figure 1-21). Counterrotating (CR) designs offer the best performance. Propellers impart swirl in the airstream as they rotate to compress the air

behind them. Since the most efficient thrust is a force that is parallel tc the forward motion of the aircraft, some thrust — and therefore, some efficiency — is lost to swirl. The opposite rotation of the CR propfan's rear propeller removes most of the swirl from the thrust of the front propeller, allowing a 'straighter' overall thrusting force, greater efficiency, and subsequently, better fuel consumption. A CR configuration also imposes lower gyroscopic loads on the engine mount, since the rotation of one propeller balances the other.

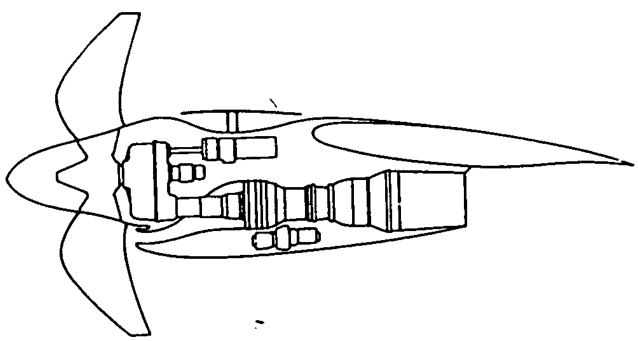


Figure 1-20 — Single-rotation Propfan

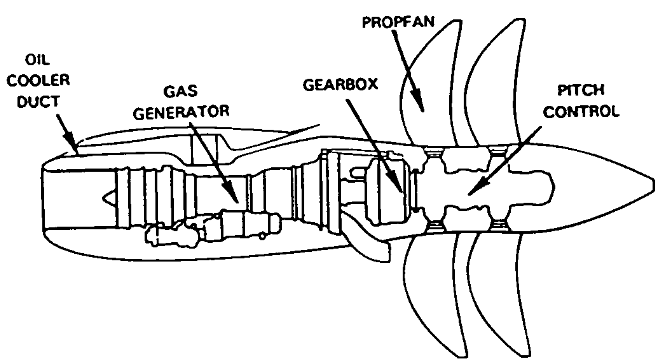


Figure 1-21 — Counterrotation Propfan

A major difference between propfan configurations is in their method of driving the propellers. Conventional turboprops have gearboxes that allow the gas generator to rotate rapidly while the propeller turns at its slower and most efficient speed. However, it is possible to design a propfan without a gearbox; the propfan blades can be driven directly by turbine exit gas if the fan blades are themselves attached to turbine blades in the gaspath — somewhat like a free turbine design — which precludes the need for reduction gearing.

Advanced Ducted Engines

Some mention should be made of concepts known as advanced ducted engines. These designs are essentially turbofans with large, swept fan blades that have pitch control and reduction gearing similar to propfans, but the fans are enclosed in ducts like turbofans (Figure 1-22). For comparison, the bypass ratio of modern turbofans is from 5:1 to 6:1, that of these ducted engines from 15:1 to 25:1, while propfans approach 90:1. Currently the advanced ducted engines have several names according to differences in configuration. There are two basic designs: one with a geared, variable pitch, single-propeller fan, and the other with counterrotating blades with pitch control. The pitch control of both designs can be used for reverse thrust like some turboprops. Advanced ducted engines may offer substantially better fuel consumption than the modern turbofans now in service. They cannot achieve the full fuel consumption potential of propfans, which have the highest bypass ratios without the added weight and drag of a duct. However, advanced ducted engines can fly at turbofan speeds without penalty, which is an advantage to long-range aircraft, whereas propfans are most economical for short to medium range aircraft.

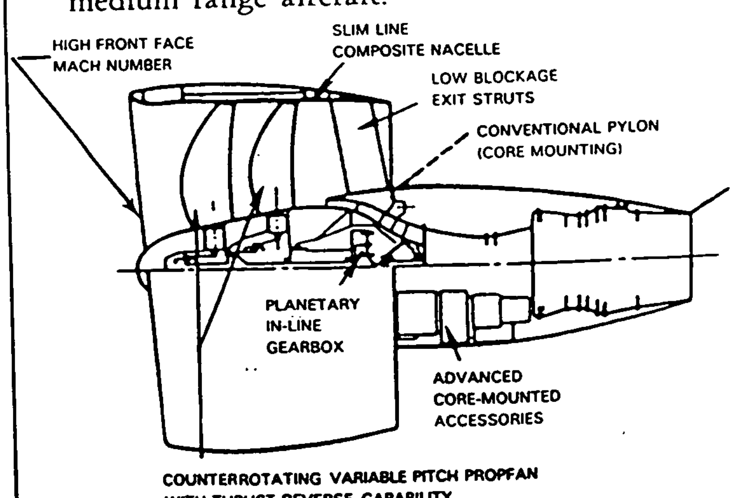


Figure 1-22 — Advanced Ducted Engines

The acoustic insulation and containment capability of the duct allows the engine to be wing mounted, a configuration that facilitates aircraft balance. The new ducted engines will also keep far field noise levels in compliance with Federal Aviation Administration and local airport regulations; passenger noise comfort will be equivalent to that of present turbofan aircraft.

Other Reaction Engines

Ramjets, pulsejets, and rocket motors are members of the reaction engine family, even though they are not classified as gas turbine engines. Because such engines are used to propel aircraft, spacecraft, and missiles, they are of interest to jet engine students who should know the nature of these powerplants. Brief descriptions of these engine types are included here.

Ramjet

The most simple jet engine is the ramjet, which has no moving parts (Figure 1-23). Essentially, a ramjet is only a large, open-ended pipe with a fuel injection and fuel metering system. As its name implies, a ramjet relies entirely upon ram effect to build up the pressure of the air entering the engine to the amount that will enable the engine to operate. This is done by accelerating the ramjet to a speed at which the shape of its air inlet can help compress the air for efficient combustion. This is just as a gas turbine on an airplane operates, but forward speed and ram effect supplants the rotating compressor of the gas turbine. Hence, a ramjet must be carried aloft and accelerated to operating speed by some means other than its own thrust. It may ride 'piggyback' on a rocket to operational altitude, or it may be borne to the proper height and speed as a dropable external store on a conventional airplane. Once they commence to operate by themselves, ramjets function on the jet propulsion principle, just like any other reaction engine.

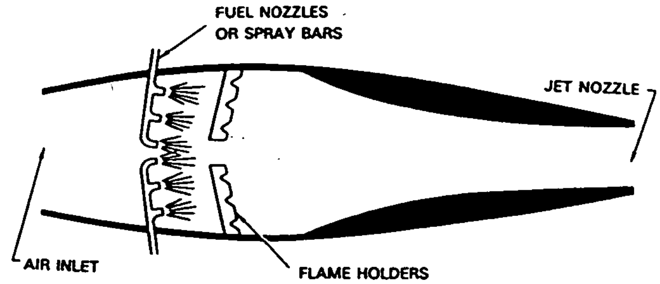


Figure 1-23 — Ramjet Engine

Pulsejet

A pulsejet (Figure 1-24) is a ramjet with an air inlet that has a set of shutters that are spring-loaded to remain in the closed position. After a pulsejet engine is launched, ram air pressure forces the shutters to open, fuel is injected into the combustion chamber, and is burned. Ignition, however, is intermittent, timed to go on and off as the shutters open and close. As soon as the pressure in the combustion chamber equals the ram air pressure, the shutters close. The gases produced by combustion are forced out the jet nozzle at the rear of the engine by the pressure that has built up within the chamber. The acceleration of the gases through the nozzle generates thrust. When the pressure drops in the combustion chamber, the shutters open again, admitting more air, and the cycle repeats itself with great rapidity. Pulsejets can be started and operated at considerably lower speed than ramjets, and it is possible to design a pulsejet that does not need a high intial velocity to be started. The V-1 "Buzz Bomb" of World War II was powered by a pulsejet.

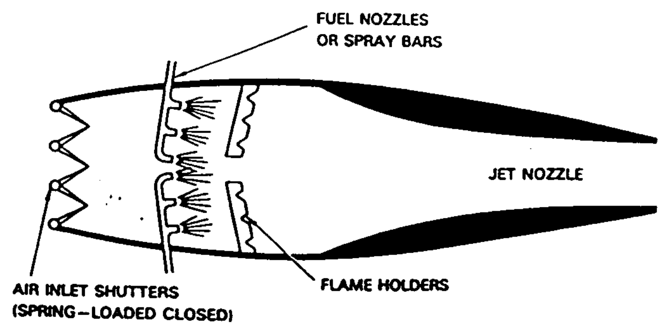


Figure 1-24 — Pulsejet Engine

Rocket Engine

A rocket engine operates on the jet propulsion principle, and carries its fuel and an oxidizer to burn with the fuel either in the rocket, itself, or aboard the vehicle that the rocket propels. Therefore, unlike ramjets, pulsejets, and gas turbine engines, a rocket engine is not an air-breathing engine, and, because of this, can operate in complete independence of the outside atmosphere, as in the airlessness of outer space. The fuel for a rocket engine and the oxidizing agent required for combustion, together constitute the propellant. Solid fuel engines carry the propellant in the combustion chamber (Figure 1-25). Fuel and oxidizer are mixed together in solid form and remain packed together until they are burned. Liquid fuel engines carry the propellant separately in tanks (Figure 1-26). Liquid oxygen is stored in one fuel tank, and a liquid fuel, usually kerosene or hydrogen, is stored in another tank. The fuel and oxidizer are pumped to the combustion chamber and are ignited. Both solid and liquid propellant engines consist primarily of the combustion chamber and a jet nozzle through which the gases of combustion are expelled at high velocity to develop thrust.

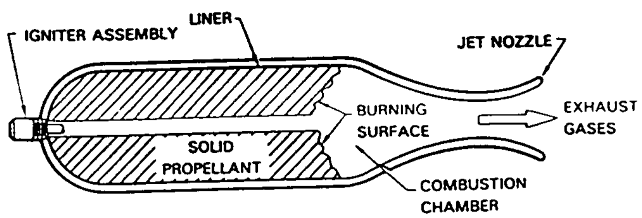


Figure 1-25 — Solid-Fuel Rocket Engine

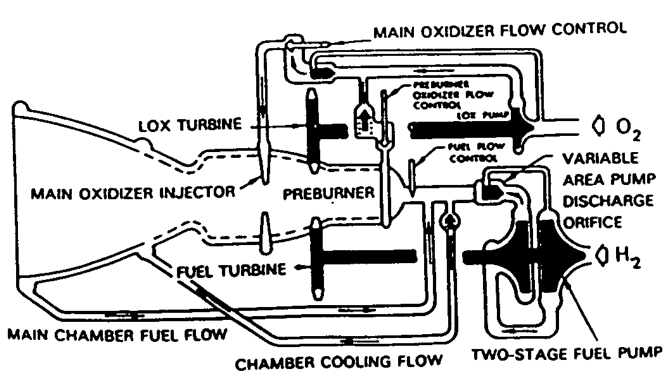


Figure 1-26 — Liquid-Fuel Rocket Engine

Hybrid Ram-Rocket Engines

Hybrid versions of ramjets and rocket engines are being considered by aerospace scientists as potential power for supersonic or hypersonic aircraft (both commercial and military). Another application for hybrid engines is in 'single-stage-to-orbit' space vehicles: spacecraft that take off from conventional runways or from launch pads without using expendable rocket stages that are jettisoned when out of fuel. Currently, spacecraft must carry their own oxidizer, but a successful hybrid ramjet-rocket would use air in the atmosphere until the vehicle reaches an altitude where the air is too thin, whereupon the rocket engine would begin to operate by using a store of liquid oxygen to attain orbit. Since the vehicle uses atmospheric air for part of its flight, much weight in oxidizer is saved, making it possible to design a vehicle that can take off and land horizontally like conventional airplanes. Besides making flight operations more convenient, such a design would also reduce expendable materials (such as jettisoned rocket stages).

A hybrid ramjet-rocket must overcome several obstacles that arise during hypersonic flight (hypersonic speed is about five times the speed of sound). At high Mach numbers, the protrusion of an engine pod creates critical drag problems. The higher the speed, the higher must be the ratio of the nozzle exit area relative to the inlet area, which means that engine performance gains can be lost to aerodynamic drag. One solution to this problem is to design the engine to be totally integrated into the airframe of the ship in order to save weight. The underside of the aircraft is contoured to help compress the air, and thus forms part of the engine intake, and the airframe behind the engine flares out to form part of the exhaust nozzle. The requirements of hypersonic operation are satisfied with minimal affect on weight and drag.

Internal engine heat is another challenge. Air for combustion must be slowed down inside the engine to prevent the flame from being blown out, but the braking effect of the engine inlet

converts velocity energy to heat energy, which can affect the combustion process. Additionally, the internal temperature can melt engine parts. Finally, the shock wave created when the air passes below the sound barrier is a problem that disrupts the airflow in the engine.

The supersonic-combustion ramjet — or simply 'scramjet' — is designed to overcome the problems of internal heat and shock. The scramjet would use hydrogen as a fuel, since hydrogen can burn within a fast airstream — the air would not have to be slowed as much as it would be if conventional fuels were used. New materials, aerodynamic concepts, and variable airflow geometry controlled by high-speed computers are other factors that will go into the making of a scramjet.

As mentioned earlier, ramjets need considerable forward speed before their inlets can compress the air enough to sustain combustion. Any ramjet design will need another form of propulsion to enable it to reach ramjet speeds. It may be possible to design a ramjet with a gas turbine section that would allow the aircraft to take off and accelerate to ramjet speeds, and to allow the aircraft to operate at the slow speeds of approach, holding patterns, go-arounds, and diverted landings to alternative airports. The gas turbine would share the same inlet and exhaust as the ramjet, but would be bypassed and deactivated when the ramjet could operate. For ramjet-rockets meant to achieve Earth orbit, a rocket engine could replace the gas turbine. A 'spaceplane' could take off with a rocket engine, switch to the scramjet at high speeds, then at high altitudes switch back to the rocket engine to attain orbit. The scramjet-rocket would reduce some of the need for liquid oxygen, allowing the spacecraft to carry wings for reusability and conventional landing and takeoff.

Note that all of the concepts discussed here are hybrids of the conventional propulsion systems — they are neither jet, ramjet, nor rocket engine, but instead are integrated designs meant to save aircraft weight and fuel load for hypersonic flight. Although these concepts are only in the design stage, the physical realities of hypersonic flight will probably dictate the use of some version of them in the future.

Compressible flow in gases

Compressible flow refers to flow at velocities that are comparable to, or exceed, the speed of sound. The compressibility is relevant because at such velocities the variations in density that occur as the fluid moves from place to place cannot be ignored.

Suppose that the fluid is a gas at a low enough pressure for the ideal equation of state, equation (118), to apply and that its thermal conductivity is so poor that the compressions and rarefactions undergone by each element of the gas may be treated as adiabatic (see above). In this case, it follows from equation (120) that the change of density accompanying any small change in pressure, $d\,p$, is such that

$$\rho^{-1} dp = \left(\frac{\gamma}{\gamma - 1}\right) d\left(\frac{p}{\rho}\right). \qquad (140)$$

This makes it possible to integrate the right-hand side of equation (131), and one thereby arrives at a version of Bernoulli's law for a steady compressible flow of gases which states that

$$\left(\frac{\gamma p}{(\gamma - 1)\rho}\right) + \frac{v^2}{2} + gz \qquad (141)$$

is constant along a streamline. An equivalent statement is that

$$\frac{C_p T}{M} + \frac{v^2}{2} + gz \qquad (142)$$

is constant along a streamline. It is worth noting that, when a gas flows through a nozzle or through a shock front (see below), the flow, though adiabatic, may not be reversible in the thermodynamic sense. Thus the entropy of the gas is not necessarily constant in such flow, and as a consequence the application of equation (120) is open to question. Fortunately, the result expressed by (141) or (142) can be established by arguments that do not involve integration of (131). It is valid for steady adiabatic flow whether this is reversible or not.

Bernoulli's law in the form of (142) may be used to estimate the variation of temperature with height in the Earth's atmosphere . Even on the calmest day the atmosphere is normally in motion because convection currents (see below Convection) are set up by heat derived from sunlight that is released at the Earth's surface. The currents are indeed adiabatic to a good approximation, and their velocity is generally small enough for the term v^2 in (142) to be negligible. One can

therefore deduce without more ado that the temperature of the atmosphere should fall off in a linear fashion—*i.e.*, that

$$T(z) = T(0) - \beta z = T(0) - \left(\frac{Mg}{C_p}\right)z. \qquad (143)$$

Here β is used to represent the temperature lapse rate, and the value suggested for this quantity, $(M\,g/C_p)$, is close to 10° C per kilometre for dry air.

This prediction is not exactly fulfilled in practice. Within the troposphere (*i.e.*, to the heights of about 10 kilometres to which convection currents extend), the mean temperature does decrease with height in a linear fashion, but β is only about 6.5° C per kilometre. It is the water vapour in the atmosphere, which condenses as the air rises and cools, that lowers the lapse rate to this value by increasing the effective value of C_p. The fact that the lapse rate is smaller for moist air than for dry air means that a stream of moist air which passes over a mountain range and which deposits its moisture as rain or snow at the summit is warmer when it descends to sea level on the other side of the range than it was when it started. The foehn wind of the Alps owes its warmth to this effect.

The variation of the pressure of the atmosphere with height may be estimated in terms of β, using the equation

$$p(z) = p(0)\left[1 - \frac{\beta z}{T(0)}\right]^{Mg/R\beta}. \qquad (144)$$

This is obtained by integration of (123), using (118) and (143).

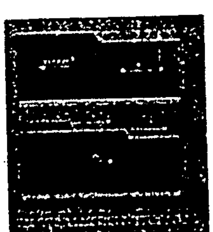


Figure 6: Steps on the surface of shallow water.

In the form of equation (141), Bernoulli's law may be used to calculate the speed of sound in gases. The argument is directly analogous to the one applied in the previous section to waves on shallow water—and, indeed, the diagrams in Figure 6 can serve to illustrate the argument here too, if they are regarded as plots of gas density (or else of pressure or temperature, which go hand in hand with density in adiabatic flow) versus position. The results of the argument will be stated without proof. If there exists an infinitesimal step in the density of the gas, it will remain stationary provided that the gas flows uniformly through it toward the region of higher density, with a velocity

$$V_s = \sqrt{\left(\frac{\gamma p}{\rho}\right)}. \qquad (145)$$

If the gas is stationary, then (145) describes the velocity with which the step moves. It also describes the speed of propagation of the sort of undulatory variation of density that constitutes a sound wave of fixed frequency or pitch. Because the speed of sound is independent of pitch, sound waves, like waves on shallow water, are nondispersive. This is just as well. It is only because there is no dispersion that one can understand the words of a distant speaker or listen to a symphony orchestra with pleasure from the back of an auditorium as well as from the front.

It should be noted that the formula for the speed of sound in gases may be proved in other ways, and Newton came close to it a century before Bernoulli's time. However, because Newton failed to appreciate the distinction between adiabatic and isothermal flow, his answer lacked the factor γ occurring in (145). The first person to correct this error was Pierre-Simon Laplace.

The above statements apply to density steps or undulations, the amplitude of which is infinitesimal, and they need some modification if the amplitude is large. In the first place it is found, as for waves on shallow water and for very much the same reasons, that, where two small density steps are moving parallel to one another, the second is bound to catch up with the first. It follows that, if there exists a propagating region in which the density rises in a continuous fashion from ρ to ρ', where $(\rho' - \rho)$ is not necessarily small, then the width of this region is bound to diminish as time passes. Ultimately a shock front develops over which the density—and hence the pressure and temperature—rises almost discontinuously. There are processes within the shock front, vaguely analogous on the molecular scale to the foaming of a breaking water wave, by which energy is dissipated as heat. The speed of propagation, V_{sh}, of a shock front in a gas that is stationary in front of it may be expressed in terms of V_s and V_s', the velocities of small-amplitude sound waves in front of the shock and behind it, respectively, by the equation

$$2V_{\mathrm{sh}}^2 = \left[\frac{(\gamma+1)\rho'}{\gamma\rho}\right] V_s'^2 + V_s^2. \qquad (146)$$

Thus, if the shock is a strong one ($\rho' \gg \rho$), V_{sh} may be significantly greater than both V_s and V_s'.

Even the gentlest sound wave, in which density and pressure initially oscillate in a smooth and sinusoidal fashion, develops into a succession of weak shock fronts in time. More noticeable shock fronts are a feature of the flow of gases at supersonic speeds through the nozzles of jet engines and accompany projectiles that are moving through stationary air at supersonic speeds. In certain circumstances when a supersonic aircraft is following a curved path, the accompanying shock wave may accidentally reinforce itself in places and thereby become offensively noticeable as a "sonic boom," which may break windowpanes and cause other damage. Strong shock fronts also occur immediately after explosions, of course, and when windowpanes are

broken by an explosion, the broken glass tends to fall outward rather than inward. Such is the case because the glass is sucked out by the relatively low density and pressure that succeed the shock itself.

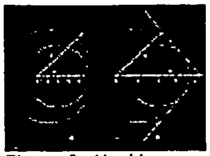


Figure 8: *Mach's construction.* (A) Source speed *U* less than speed of sound ...

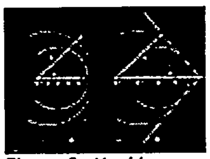


Figure 8: *Mach's construction.* (A) Source speed *U* less than speed of sound ...

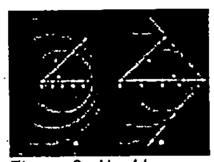


Figure 8: *Mach's construction.* (A) Source speed *U* less than speed of sound ...

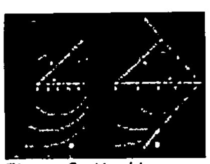


Figure 8: *Mach's construction.* (A) Source speed *U* less than speed of sound ...

The diagrams in Figure 8 show a well-known construction attributed to the Austrian physicist Ernst Mach that explains the origin of the shock front accompanying a supersonic projectile. The circular arcs in this figure represent cross sections through spherical disturbances that are spreading with speed V_s from centres (S′, S″, etc.), which mark the position of the moving source S at the time when they were emitted. If the source is something like the tip of an arrow, which disturbs the air by parting it as it travels along but which is inaudible when stationary, then each "disturbance" due to some infinitesimal displacement of the tip is a spherical shell of infinitesimal thickness within which a small radial velocity has been imparted to the air. There is an infinite number of such disturbances, overlapping one another, of which only a handful are represented in Figure 8. When the velocity of the source, U, is less than V_s (Figure 8A), the result of adding them together is the sort of steady backflow that is to be expected around a moving obstacle, and there is no sound emission in the normal sense; the source remains inaudible. When U exceeds V_s, however, the spherical disturbances reinforce one another, as Figure 8B shows, on a conical caustic surface, which makes an angle of $\sin^{-1}(U/V)$ to the line of travel of the source, and it is on this surface that a shock front is to be expected. The cone becomes sharper as the source speeds up.

(14)

Mach, Ernst

born Feb. 18, 1838 , Chirlitz-Turas, Moravia, Austrian Empire
died Feb. 19, 1916 , Haar, Ger.

Austrian physicist and philosopher who established important principles of optics, mechanics, and wave dynamics and who supported the view that all knowledge is a conceptual organization of the data of sensory experience (or observation).

Mach was educated at home until the age of 14, then went briefly to gymnasium (high school) before entering the University of Vienna at 17. He received his doctorate in physics in 1860 and taught mechanics and physics in Vienna until 1864, when he became professor of mathematics at the University of Graz. Mach's interests had already begun to turn to the psychology and physiology of sensation, although he continued to identify himself as a physicist and to conduct physical research throughout his career. During the 1860s he discovered the physiological phenomenon that has come to be called Mach's bands, the tendency of the human eye to see bright or dark bands near the boundaries between areas of sharply differing illumination.

Mach left Graz to become professor of experimental physics at the Charles University in Prague in 1867, remaining there for the next 28 years. There he conducted studies on kinesthetic sensation, the feeling associated with movement and acceleration. Between 1873 and 1893 he developed optical and photographic techniques for the measurement of sound waves and wave propagation. In 1887 he established the principles of supersonics and the Mach number—the ratio of the velocity of an object to the velocity of sound.

In *Beiträge zur Analyse der Empfindungen* (1886; *Contributions to the Analysis of the Sensations*, 1897), Mach advanced the concept that all knowledge is derived from sensation; thus, phenomena under scientific investigation can be understood only in terms of experiences, or "sensations," present in the observation of the phenomena. This view leads to the position that no statement in natural science is admissible unless it is empirically verifiable. Mach's exceptionally rigorous criteria of verifiability led him to reject such metaphysical concepts as absolute time and space, and prepared the way for the Einstein relativity theory.

Mach also proposed the physical principle, known as Mach's principle, that inertia (the tendency of a body at rest to remain at rest and of a body in motion to continue in motion in the same direction) results from a relationship of that object with all the rest of the matter in the universe. Inertia, Mach argued, applies only as a function of the interaction between one body and other bodies in the universe, even at enormous distances. Mach's inertial theories also were cited by Einstein as one of the inspirations for his theories of relativity.

Mach returned to the University of Vienna as professor of inductive philosophy in 1895, but he suffered a stroke two years later and retired from active research in 1901, when he was appointed to the Austrian parliament. He continued to lecture and write in retirement, publishing *Erkenntnis und Irrtum* ("Knowledge and Error") in 1905 and an autobiography in 1910.